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ECONOMIC ANALYSIS OF STANDARD INTERFACE MODULES FOR USE WITH THE MULTI-MISSION SPACECRAFT

VOLUME I

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FINAL

ECONOMIC ANALYSIS OF STANDARD INTERFACE MODULES FOR USE WITH THE MULTI-MISSION SPACECRAFT

VOLUME I

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NOTE OF TRANSMITTAL

This economic analysis of Standard Interface Modules (SIM) for use with the Multi-Mission Spacecraft (MMS) was performed for NASA by ECON, Inc. under Contract No. NASW-2558. The Technical Officer for this study was Mr. Rondal Crawford of NASA Headquarters. ECON, Inc. was assisted in this study by Kaman Sciences Corporation. The study evaluates the cost savings that could be obtained by the use of SIM to perform certain sensor electrical interfacing functions that have historically been an integral part of the sensor.

The study concludes that reduction in both the nonrecurring and recurring costs of this sensor interface hardware could be achieved through the use of SIM, and that the development and use of certain power conditioning and data handling SIM units is economically justified. An important conclusion of this study is that greater cost savings could be realized by the extension of the SIM concept to the planned Spacelab missions, and that further study of the use of SIM in those manned missions is warranted.

The analysis of the sensor interface functions for the MMS missions was performed by Mr. Samuel Russell of ECON, Inc. Mr. Noel Becar of Kaman Sciences Corporation was responsible for the selection of functions to be standardized, and the development of the physical characteristics of the selected SIM. The costing and economic analysis was performed by Mr. Joel Greenberg of ECON, Inc. The RCA PRICE cost estimating program was used by ECON, Inc. to estimate the costs of both the SIM and integral design concept hardware.

The principal authors of this report were Mr. Joel Greenberg, Mr. B.P. Miller and Mr. Samuel Russell of ECON, Inc., and Mr. Noel Becar of Kaman Sciences Corporation.

BPMiller

B. P. Miller
Vice President

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1. EXECUTIVE SUMMARY

This report describes the results of a preliminary technical and economic feasibility study of the use of Standarized Inter-state Modules (SIM) to perform electual interfacing functions that have historically been incorporated into sensors.

The objective of this study is to identify sensor interface functions that are capable of standarization from the set of missions planned for the NASA Multi-Mission Spacecraft (MMS) in the 1981 to 1985 time period, and to examine the cost savings that could be achieved through the replacement of nonstandard sensor interface flight hardware that might be used in these missions with SIM.

The methodology used in this study consisted of:

1. An examination of the sensor electrical interface characteristics of the MMS,
2. An analysis of the electrical interface requirements of the sensors that might be flown on MMS missions planned for 1981 through 1985,
3. The selection of the set of electrical interface functions that are capable of standardization for this mission set,
4. The definition of the hardware characteristics of these electrical interface functions for the two alternative design cases considered:
    a. Continuation of the historical practice of incorporating the electrical interface functions into the individual sensors, or
    b. Development and production of SIM,
5. Estimation of the nonrecurring and recurring costs for the cases a and b above, in order to estimate the cost savings achieved by standardization.

The results of this study indicate that a significant degree of standardization could be achieved for the sensor electrical interface functions of power conditioning and sensor data handling for the MMS missions considered. Four types of standard power conditioning modules, and two types of standard data processing modules were identified as feasible. The use of these SIM in the 31 MMS flights anticipated in the 1981-1985 time period could result in a net cost savings to NASA in the range of $17.7 million to $21.1 million. A preliminary consideration of the possible extension of the use of these specific SIM to the SMMS, Spacelab, and other missions contemplated for the same time period leads to an estimate of total net cost savings in the range of $65 million to $143.5 million through the standardization of sensor electrical interface functions. It should be noted that the estimated cost savings across the entire mission model may be understated, as the opportunities for standardization should increase as the set of missions considered is enlarged.

The results of this study indicate that the development and use of SIM with the MMS is economically attractive. However, a more important conclusion is that greater cost savings can probably be realized by the extension of this concept to the planned Spacelab missions. For this reason, NASA is urged to consider the requirements for the use of SIM as well as the economic effects of standardization across the entire mission model, as opposed to considering only the sensor electrical interface standardization possibilities for the MMS missions.

2. INTRODUCTION

Equipment standardization has been used in many industries to reduce both unit production costs as well as design and development costs for successive users. Standardization has been used with success in aircraft, automobiles, electronics, and weaponry, and has often gained acceptance within an industry as the technology associated with that industry matures.

The space sector of the aerospace industry developed during the late 1950s and 1960s with a strong inheritance of technology from the airframe and missile sectors of the aerospace industry and the electronics industry. Within the space sector standardization was first achieved in areas of launch vehicles, guidance systems, and ground based tracking radars as a result of the need for improved reliability of launch systems and the requirement of cost effectiveness. As many spacecraft were designed and built during the 1960s, a degree of standardization of electronic piece parts was achieved under the impetus of the need for improved quality control and reliability. However, full standardization was not achieved in the electronics piece parts area during that time period as a result of both the rapidly changing technology in the electronics industry, and the use of different levels of specifications by various parts of both the military and civilian space programs.

The feasibility of standardization at the subsystem and spacecraft levels has become apparent as a result of nearly twenty years of experience in the design, development, and production of spacecraft. Historically, in the spacecraft area, a degree of standardization was first achieved by the use of the block buy concept in programs such as TIROS, Transit, and the Intelsat communications satellites. In other programs such as NIMBUS, LANDSAT, Mariner, and the DOD orbiting Agena applications the basic spacecraft (consisting of the structure and support subsystems) remained relatively unchanged from mission to mission, while the sensor payloads were changed to meet the unique applications or science requirements of the mission. Although the number of spacecraft designs remains large, standardization was achieved within some programs, and some spacecraft subsystems such as thrusters, attitude control sensors, telemetry, transponders, and command receiver/decoders achieved a degree of standardization through the use of the same equipment design in several programs.

With the advent of the Space Shuttle as a new and standarized form of space transportation, NASA has placed increased emphasis on the development of standard spacecraft, subsystems and components as a means to achieve further cost effectiveness. A Low Cost Systems Office has been established at NASA Headquarters to foster and manage the development of standardized systems, software, and practices.* Within the Low Costs Systems Office, and at the NASA Goddard Space Flight Center, a major effort has been devoted to the design and development of modular standardized spacecraft for use in multiple missions. Two such standardized spacecraft are currently under design or development, namely, a Multi-Mission Spacecraft (MMS) and a Small Multi-Mission Spacecraft (SMMS).** The former is generally intended for use in that class of missions which now uses the Delta (or larger) launch vehicle, while the latter appears to be generally intended for missions which now use the Scout launch vehicle.

The design of a standardized spacecraft implies the existence of a standard set of interface requirements for the sensors or other mission peculiar equipment to be carried by the spacecraft. For a given element of mission payload, the standard interfaces will probably be both electrical and mechanical. The

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*Tischler, A.O., "Lower Space Cost Means More Space Flight." Aeronautics and Astronautics, June 1974.

**Low Cost Modules Spacecraft Description, Goddard Space Flight Center, May 1975.

mechanical interfaces, consisting of structural and thermal requirements, will be met by the adequate design of payload mounting pads, attachments, heatflow paths, and thermal control mechanisms. On the other hand, mission sensors such as TV cameras, detectors, counters and radiometers have required a wide range of voltages, data bit rates and commands. Historically, these requirements have been met by designing into each sensor the necessary power conditioning equipment, data formatters, data buffers, and command distribution circuitry to meet the unique requirement of each sensor.

The purpose of this study was to evaluate, in a preliminary way, the economic feasibility of using Standard Interface Modules (SIM), in conjunction with standardized spacecraft, to perform the electrical interfacing functions that have historically been incorporated into the sensors. The approach used in this study was to:

1. Examine the missions that are now expected to be flown in the five-year period extending from 1981 through 1985, and determine the electrical interface characteristics of the spacecraft to be used and the mission payloads (sensors) to be carried by the spacecraft. Because of time and resource constraints, and to facilitate the collection of the necessary data base on the collection and sensors, it was decided to limit this spacecraft and sensors, it was decided to limit this study to a subset of the science and applications missions contained in the May 1976 NASA Payload Model for Standard Equipment Planning. The subset of missions used in this study was selected on the basis of the anticipated availability of data concerning the characteristics of the payloads and their interface requirements. The need to know the standardized spacecraft bus interface characteristics imposed a further constraint on the subset of missions used in this study, since as of the time of this study (April to August 1976), the design specifications of the MMS have been published, but the interface characteristics of the SMMS were not available.

2. Define the mission elements which could be removed from the sensors and incorporated into SIM. This was accomplished by examining the sensor hardware in the subset of missions considered in this study for functions that could be removed from the sensors and made common to a number of missions for use as SIM.

3. Determine the physical characteristics (size, weight, construction, nature of technology used) for the candidate SIM. Estimate the quantities and schedule of SIM required to support the schedule of flights in the mission subset selected from the May 1976 NASA Payload Model.

4. Use the RCA PRICE computerized cost estimating model to estimate the nonrecurring and recurring costs under the alternative assumptions of:

   a. Development and production of the SIM (i.e., standardization)

   b. A continuation of the historical approach of incorporating peculiar power conditioning, data, and command functions in the individual sensors.

   The difference in the present value of costs between the cost streams developed under assumptions a and b above is then the preliminary estimate of the economic benefits which may be derived from the use of the SIM in the selected missions.

5. Perform a sensitivity analysis on the estimated economic benefit by varying factors such as the quantity of SIM used and their physical and/or design characteristics.

As a result of schedule and resource constraints several important factors which could have an economic impact on the use of SIM were not considered in this analysis. The factors not considered include the effect on cost of reliability improvements that may be possible with the SIM, as well as optimum inventory or block buy techniques. In addition, this study only considered the use of SIM in presently identified MMS missions. The further use of the SIM identified in this study in the SMMS, Spacelab, commercial, or foreign missions was not considered. These factors should all be considered in any further study aimed at refining the economic benefits of the use of SIM.

3. MISSION ANALYSIS AND SELECTION

3.1 Characteristics of Planned Space Operations, 1981-1985

The Space Shuttle is scheduled to become operational from NASA Kennedy Space Center during the third quarter of FY 1980. Operations from Vandenbergh Air Force Base are scheduled to commence during the second half of FY 1982. These events will signal the beginning of a new era of space transportation with the ability of the Space Shuttle, operating in conjunction with an Interim Upper Stage, to inexpensively transport a wide range of payloads to orbit. The primary operations goal for the Space Shuttle program is to provide low cost transportation to and from Earth orbit. To achieve further cost savings during the period of Space Shuttle operations, NASA is also developing standardized multimission spacecraft. These standardized multimission spacecraft are intended to effect cost savings by reducing the nonrecurring design and development costs as well as the recurring (unit production costs) by replacing the wide variety of spacecraft that in the past have been developed for each new mission. The objective of the Multi-Mission Spacecraft is to standardize to the maximum possible extent the spacecraft structure, thermal design, attitude control, communications, data handling, power, and telementry subsystems. Certain of these subsystems such as communications, power, and telemetry could be standardized in a modular fashion to provide for unique mission requirements in these areas. Thus, in the multi-mission spacecraft concept the variability, and thus the main element of new design, would be isolated to the unique mission payload or sensors.

The SIM concept provides for a further extension of this standardization by the identification of those sensor (payload) functions, such as power conditioning, data processing, signal conditioning, and command distribution, where the functional characteristics in these areas are common to many sensors.

By use of the SIM the total of the nonrecurring costs of sensor (payload) development and the recurring or production costs, for performing a set of missions, could be reduced.

The time period of 1981 to 1985 was selected for the economic analysis of the SIM on the basis of an estimate of the time that would be required to design, develop, and produce the SIM for use with the MMS. Assuming a period of further study leading to the preparation of specifications for the selected SIM, a period of 18 to 24 months should then be adequate for the design, development and initial production of flight units. Thus, given a decision to implement the SIM concept in FY 1977 (or early in FY 1978), flight qualified production units of SIM could be available for integration with flight spacecraft during FY 1980.

Figure 3.1 is the composite payload planning model which has been used as the basis for estimating the demand for SIM. This composite payload planning model is derived from two sources:

1. The OSS, OA, and OAST missions are obtained from the NASA Payload Model for Standard Equipment Planning, dated May 1976.

2. The Applications (other Gov't) missions are obtained from the Interim NASA Payload Model for Planning Purposes, dated March 8, 1976. These missions represent the operational derivatives of the OA TIROS, STORMSAT, and LANDSAT missions.

The missions that are now considered to be candidates for the use of the MMS are circled in Figure 3.1. Figure 3.2, derived from Figure 3.1, is a mission model summary and illustrates the quantity of MMS flights considered in this study as a function of time. The first MMS mission shown is scheduled for FY 1980 (Solar Maximum Mission and Technology Demonstration Satellite). For the purpose of this study, the FY 1980 flights of the MMS are not considered as targets for the use of the SIM, in order to allow the necessary time for further evaluation, decision making, design, development, and production of the SIM. Considering the scheduling

	Fiscal Years								
	77	78	79	80	81	82	83	84	85
OSS									
Explorers (HEATE)	X	XX	X	XX	XX	XX	XX	XX	XX
Solar Maximum Mission				X			X		X
Gravity Probe B						X	X		
Space Telescope						X	X		
HEAO BLK II								X	XX
Solar Observatory									X
Spacelab				X	X	X	X	X	X
HEAO	X	X		X					
Mars Polar Orbiter								X	
Mariner Jupiter Orbiter									X
Lunar Polar Orbiter						X			
Pioneer Jupiter Orbiter Probe						X			
Halley Flyby									X
Jupiter Swingby Out-of-Elip						X			
Pioneer Saturn/Uranus/Titan							X	X	X
Venus Orbiter Imaging Radar							X		
Mariner Mercury Orbiter							X		
Out-of-Elip Solar Observ								X	
MJS '77	XX								
Pioneer Venus		XX							
Vestibular Function Research					X				
Life Science Carry Ons (Spacelab)					X	XX	XX	XXX	XXX
Biomedical Experiment Satellite (BESS)						XX	X	X	XX
OA									
Storm Sat A/B					X			X	
Radiation Budget						XX			
Tiros N, N', O		X			X		X		
Seasat A, B, C		X				X			X
Geological Satellite				X					
Nimbus G			X						
Environmental Monitoring Sat A/B						X		X	
HCMM/SAGE		X	X						
Landsat C, D, D', E, E'	X				X	X		X	X
Synchronous Earth Observations Sat A/B									X
Technology Demonstration				X					
Earth Survey Satellite							X		
Gravsat A								X	
Search and Rescue					X				
Disaster Warning Satellite System						X	X		
Mobile System Commun. Sat							X		
Earth Viewing Applications Laboratory						X	X	XX	XX
Space Processing Labs					X	X	X	XX	XX
OAST									
Long Duration Exposure Facility				X		X		X	
SPHINX Satellite					X				
Spacelab Payloads (1/4)				2	3	4	5	6	6
APPLICATIONS (OTHER GOV'T)									
NOAA A-G		X	X	X	X	X	X	X	
GOES (NOAA) A-G	X	X		X	X		X	X	
EARTH RESOURCES						X	X	X	X

NOTE: O = MMS Missions

Figure 3.1 Composite Payload Planning Model

		Fiscal Years						
		80	81	82	83	84	85	Total
• HEATE			1					1
• Solar Maximum Mission		1			1		1	3
• Space Telescope					1			1
• HEAO BLK II						1	2	3
• Solar Observatory							1	1
• Stormsat A/B			1			1		2
• Tiros N', O			1		1			2
• Seasat B, C				1			1	2
• Environmental Monitoring Sat A/B				1		1		2
• Landsat D, D', E, E'			1	1		1	1	4
• SEOS							1	1
• Technology Demonstration		1						1
• Earth Survey Satellite					1			1
• NOAA E, F, G	Applications			1	1	1		3
• GOES (NOAA) F, G	Applications				1	1		2
• EARTH RESOURCES	Applications			1	1	1	1	4
								33

Figure 3.2 Mission Model Summary

factors discussed above, the MMS missions scheduled for FY 1981 are considered to be realistic targets for the initial use of the SIM. In order to bound the economic analysis, a conservative estimate of the useful life of technology of the SIM and its applications of five years has been made. This estimate is based upon experience with the rate of change of electronics piece parts and sensor technology in space programs over the past fifteen years. The actual need date for second generation SIM will be determined by the rate of change in these areas in the early 1980s, as well as by the electrical interface characteristics of the sensors developed for missions to be flown after 1985. While this five-year life implies that a second generation of SIM may be required after 1985, it is likely that the second generation SIM will contain significant technical inheritance from the first generation, thus reducing the nonrecurring costs of the second generation units.

3.2 The Multi-Mission Modular Spacecraft

The Multi-Mission Modular Spacecraft (MMS) is the result of a six-year NASA Goddard Space Flight Center study on standardized approach to supporting many flight missions and which is compatible with both expendable and Space Shuttle vehicles.

Specific potential cost savings aspects of the MSS design are:

1. Maximum use of standard components
2. Standardized subsystem modules for a variety of mission classes
3. Exploitation of shuttle capability for resupply and retrieval
4. Standardizided flight and ground software and utilization of standard ground support and operational systems.

An exploded view of the MMS is shown in Figure 3.3. Of particular interest to this study is the electrical interface

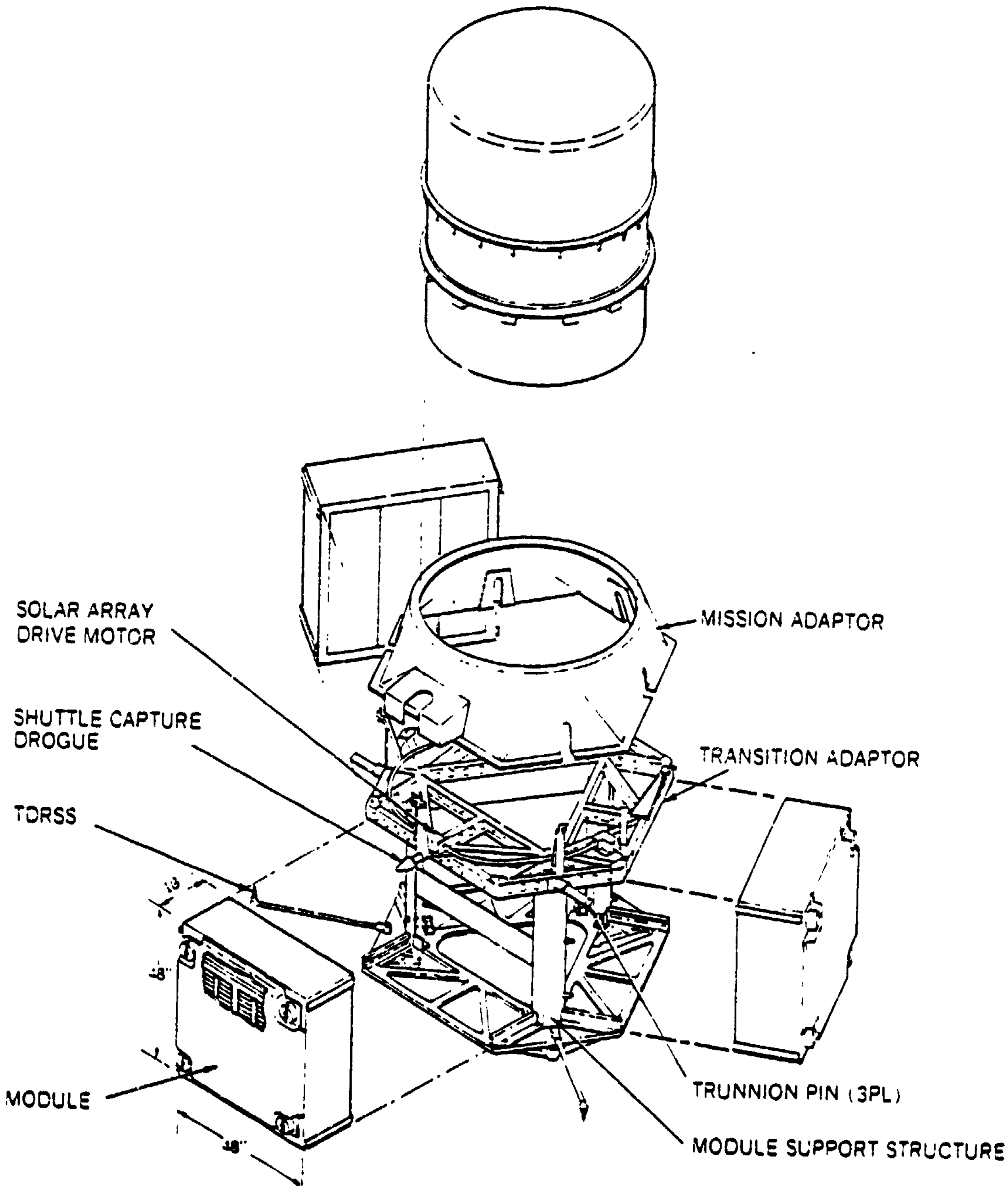

Figure 3.3 Multi-Mission Modular Spacecraft (Source: Low Cost Modular Spacecraft Description, NASA, GSFC, May 1975)

between the Power System and Command and Data Handling System with different mission sensors. Detailed below are specifics of these interfaces.

3.2.1 Power Interface

Power is supplied by a solar array and rechargable battery system. The solar array provides power through an unregulated bus which is connected to the spacecraft, instruments, and batteries. The batteries supply the bus while the spacecraft is in eclipse and also during peak overload periods.

Table 3.1 defines the power interface at the mission payload distribution point. Although designated an unregulated supply, the specification imposes many of the restrictions on the user (payload sensors) that would be characteristic of a highly regulated supply. Thus the system imposes on the user the penalties of both regulated and unregulated systems, namely:

1. Imposition of strict limitations of user perturbations to the power bus
2. Poor voltage regulation.

3.2.2 Command and Telemetry Interfaces

Command and Telemetry interfaces to the MMS are provided through a Remote Interface Unit (RIU). Command signals are delivered either as Discrete Commands (pulse) or as Serial Magnitude Commands (serial digital bit stream). Each RIU contains a command decoder which has 64 Discrete Command outputs and 8 Serial Magnitude Command outputs. Discrete Commands are provided as single-ended switch closures to signal ground when selected (active state). The normal state of the switch is open (inactive state). Table 3.2 provides details of the command interfaces.

The RIU will contain a multiplexer having 64 inputs that can be used for analog, bilevel, and serial digital signals as detailed in Table 3.3. The signal handling capability will allow the sensor to use any input for analogs, any input for

Table 3.1 MMS Power Interface

Voltage: Nominal +28 $\pm$ 7 VDC

Impedance:	0.07 ohms	- 1 Hz to 1 KHz
	0.10 ohms	- 1 KHz to 20 KHz
	0.30 ohms	- 20 KHz to 150 KHz
	0.50 ohms	- 150 KHz to 10 MHz

Power Supply Transients:
a) Normal transients: within + 21 to + 35 VDC.
b) Abnormal transients (system fault) within 0 to + 40 VDC.

Ripple, Supply Output: $\leq$ 500 MV, p to p, 1 Hz to 10 MHz.

Turn-On Transients: a) for loads under 50 watts: not to exceed 300 percent of the maximum steady state current. b) for loads greater than 50 watts: not to exceed 200 percent of the maximum steady state current.

Maximum Duration: 50 milliseconds

Maximum rate of change of current: 20 ma./microsecond

Turn-Off Transients: peak voltage transients generated on the power lines by inductive effects of the load to be within zero to + 40 volts.

Operational Transients: not to exceed 125 percent of the maximum peak operational current.

Maximum Duration: 50 milliseconds

Maximum rate of change of current: 20 ma./microsecond

Reflected Ripple Current: not to exceed 5 percent of the steady state current drawn. The fundamental frequency of load current ripple shall not exceed 100 KHz.

Table 3.2 MMS Command Interface

Discrete Commands (63 per remote unit):	
Inactive State:	Output impedance = 1 megohm User termination 1 K ohm to user Signal Power: Typical + 5.0 volts Maximum + 30.0 volts
Active State:	
Duration	6.5 to 7 millisec .5 V maximum @ 20 ma.
Relay Drive Pulse:	
Pulse Duration	6.5 to 7 millisec
Active state:	+ 28 ± 2 V @ 20 ma. ± 4 V @ 40 ma max.
Inactive state:	0 to + .5 V @ 0 ma.
Serial Magnitude Commands (8 per remote unit):	
Clock	16 pulses at 256 KHz
Gate	72 μsec wide Envelope is 16 clock pulses beginning 10.7 μsec before the first clock pulse. Active and inactive levels and user termination are the same as for discrete commands.
Command Word	16 bits serial NRZ-L data with bit transitions occurring 1 microsecond after the trailing edge of the clock pulse. The user should use the trailing edge for each clock pulse for shifting.

Table 3.3 MMS Telemetry Interface

Analog Inputs (digitized to 8 bits in the RIU):	
Range	0 to +5V
User Impedance Source	5K ohms maximum
Accuracy	+ 20 MV
Selective Conditioning	16 inputs will be capable of conditioning passive transducers with 1.0 ma constant current at time of sampling.
Bilevel Digital Inputs:	
Logical "1"	+3.5 to +15 VDC
Logical "0"	-0.8 to +0.8 VDC
User Z Source	5K ohms maximum
Serial Digital Inputs (8 bits/word):	
Clock (Mux Output)	8 clock pulses at 256 KHz
Gate (Mux Output)	12 clock periods (47 μsec), beginning 4 clock periods before 8 clock pulses. Active and inactive levels and user termination are the same as for discrete commands.
Input Data	The serial data must be NR and each bit transition should coincide with trailing edge of clock.
Logical "1"	+3.5 to +12 volts
Logical "0"	-0.8 to +0.8 volts
User Z Source	500 ohms maximum

All inputs of the RIU multiplexer will have an input impedance of 10 megohms minimum in the normal mode and 1 megohm minimum during sampling.

Under fault conditions, the sensor shall not output on telemetry output lines any voltage exceeding +35 V or -15 V.

The sensor shall be able to accept, as a fault condition of the interfacing spacecraft telemetry system, the application of a voltage of +35 V to -15 V to any of the telemetry outputs for any length of time without affecting the performance of the sensor.

bilevel (in groups of 8), and any of 16 inputs for serial digital signals. The quantity of multiplexer inputs may be expanded in groups of 64 up to a total of 512 by means of expander units.

3.3 Planned MMS Missions

The missions selected for the SIM study are described in Table 3.4. A major task in the study was to define sensor characteristics for these missions. However, since none of the missions are to fly before 1980 it is difficult to obtain documentation sufficiently detailed for the SIM evaluation. Particularly lacking were details on power command and telemetry interfaces. To obtain the best information possible, interviews were conducted with the personnel most directly associated with sensor developments. The following is a definition of sensor characteristics for the selected missions and information source references.

3.3.1 STORMSAT

Discussions were held with Ms. Barbara Walton, STORMSAT Assistant Project Manager, Walt Rasking, Study Director for the Advanced Atmospheric Sounding and Imaging Radiometer (AASIR) and Mr. Jim Shiue, Study Manager for the Microwave Atmospheric Sounding Radiometer (MASR).

Detailed information was obtainted on the AASIR sensor telescope. However, a detailed study is just now being started at GSFC to define the data handling system. Thus, data and other interfaces are not well defined at this time. Also, detailed power requirements were not obtainable.

The MASR is much more poorly defined from an overall viewpoint. However, it probably will use much of the same technology used on the NIMBUS 6; thus, the data handling requirements are defined by similarity to the NIMBUS experiments. Details of the AASIR and MASR sensors are presented in Tables 3.5 and 3.6.

Table 3.4 MMS Missions Selected for Detailed SIM Investigation

Mission	STORMSAT	LANDSAT D & E	HEATE
Number of Flights	One	Two	One
Launch	1982	1979	Circa 1981
Flight Duration	2 years	2 years	
Launch Site & Vehicle	KSC, Shuttle Launcher	WTR, Delta 3910 or Shuttle.	Cape Canaveral, Delta Launch
Orbit	Geosynchronous	705 Km circular, 98.2° inclination, sun synchronous	500 Km circular, 30° inclination
Sensors	Advanced Atmospheric Sounding & Imaging Radiometer (AASIR), Microwave Atmospheric Sounding & Imaging Radiometer (MASR)	Multi-Spectral Scanner (MSS), Thematic Mapper (TM), Synthetic Aperture Radar (SAR)	Many sensors proposed. Final sensor complement to be defined.
Description	Geostationary R&D satellite for survey & monitoring of storm systems. Provides temperature and moisture profiling, rain mapping, wind vectors, etc.	Designed to extend the capability of the Earth Observation Satellite program. Also provides continuity in data for on-going earth observation programs.	Full sky survey of X-Ray and γ-Ray transient phenomena.
Information Sources	Discussions with Jack Over, STORMSAT Study Manager; Barbara Walton, STORMSAT Asst. Study Manager	M. Maxwell, GSFC., MSS final report, Hughes Aircraft Co. TM - GSFC package to industry - March 1976	J. Holtz, Explorer Program Manager for OSS, Marius Weinreb, GSFC

Table 3.4 MMS Missions Selected for Detailed SIM Investigation (Continued)

Mission	GRE	TIROS N/O	SEASAT
Number of Flights	One	Three	One
Launch	1979-80	Tiros N - 1978 Tiros O - 1981, 1984	
Flight Duration	one year	two years	
Launch Site & Vehicle	Shuttle (3rd Engineering Test Flight), KSC Launch	WTR, Delta	
Orbit	500 Km near circular, 28.5 inclination	900n. miles, sun synchronous	
Sensors	Energetic Gamma Ray Experiment Telescope Instrument	Advanced Very High Resolution Radiometer, TOVS, DCS, SEM	SMMR, SCAT, VIRR, SAR, ALT
Description	Study and mapping of transient gamma ray phenomena	Fourth generation meteorological satellite	Global survey of sea/air interface: wind, sea state, sea height, temperature, humidity, etc.
Information Sources	Execution Phase Project Plan for Gamma Ray Explorer, November, 1975.	RCA AstroElectronics Division	JPL

Table 3.5 Advanced Atmospheric Sounding and Imaging Radiometer (AASIR)

PHYSICAL CHARACTERISTICS:

Size:	1.68 meters long	
	.64 meter diameter	
	.61 meter sunshade	
Weight:	113 Kg	

SENSOR DESCRIPTION:

AASIR is an object plane scanning telescope providing visual and IR imagery and IR sounding data.

The number of physical sensors are as follows:

Imagery:	visible:	21 discrete element linear array
	3.7 μ:	6 elements
	11.1 μ:	3 elements
Sounding:	IR:	18 elements

Imaging channels are step scanned at the image plane to provide the increased resolution over that of the sounding channels without a proportional increase in the number of physical sensing elements.

INSTRUMENTATION DESCRIPTION:

The AASIR Instrument is designed to provide maximum flexibility in operating mode and scan format. However, AASIR is primarily a survey instrument and there are no plans to point the AASIR toward a designated area. Commands therefore are expected to be discrete on/off types. The present plan for data processing is to convert sounding and image data to digital form and buffer within the sensor.

POWER REQUIREMENTS:

Specific voltages and voltage regulations requirements have not been established within the AASIR. Overall power required is 47 watts.

COMMAND INTERFACE:

Undefined. All commands are probably discrete on/off--either pulse or relay closure. No magnitude commands have been identified. Study now underway will define command requirements.

DATA/TM INTERFACE:

Internally multiplexed data. Output is 10.7 Megabit/second serial data stream. Engineering and housekeeping data requirements are undefined at this time but are expected to represent a very small fraction of the total data bandwidth required.

Table 3.6 Microwave Atmospheric Sounding Radiometer (MASR)

PHYSICAL CHARACTERISTICS:

Major component of MASR is a solid aperture offset parabolic antenna between 3.0 and 4.4 meters in diameter (4.4 meters is limit of shuttle bay envelope). Weight is presently undefined.

SENSOR DESCRIPTION:

The MASR is a passive microwave radiometer to be used for rain mapping. The MASR will sound 10 channels around an O_2 absorption band at 118 GHz, and another 10 channels around an H_2O band at 183 GHz. The MASR will raster scan a square area either 750 Km or 1500 Km on a side by means of a doubly articulated gimbal drive. IFOV is 28 end 18 Km for 118 GHz and 183 GHz respectively, assuming a 4.4 meter dish. Sounding is to cover altitude range from 0-30 Km.

INSTRUMENT DESCRIPTION:

Areas not well defined at this time are the antenna, feed, gimbal drive and receiver front end. Receiver IF and successive stages are quite conventional, probably very similar to Nimbus E&F Microwave radiometers. Radiometric measurements are made through synchronous demodulation of signal and sample-and-hold detection with a 10-bit A/D conversion.

POWER INTERFACE:

Voltages and regulation requirements are not defined. Power consumption is estimated at 100 watts total:

Gimbal Drive:	50 watts
Receiver electronics and Calibration Source:	50 watts

COMMAND AND CONTROL INTERFACE:

MASR will be designated to point in specific areas by ground command. There may be from 5 to 20 operating modes and it is presently foreseen that the scan pattern may be operated by software in the MMS central computer. Command rates across the MMS/MASR interface might run 10-50 bits/second for scan definition along with other discrete commands defining the operating modes.

DATA/TM INTERFACE:

Sensor data rate is based on an IFOC dwell time of 1 second.

Total Data Sampling rate is 1 kilobit/second.

It is estimated that the engineering and housekeeping data will also be about 1 kilobit.

3.3.2 LANDSAT D and E

One of the difficulties in defining the sensor characteristics for LANDSAT D and E are that none of these missions are well defined. However, in discussions held with Marvin Maxwell of GSFC, it was determined that the best estimate of these missions is as follows: LANDSAT D is a follow-on to the current missions. LANDSAT D will carry a Multi-spectral Scanner (MSS) and a Thematic Mapper (TM). Details of the MSS are given in Table 3.7. Information was obtained from the Hughes Final Report on the MSS. Details of the Thematic Mapper given in Table 3.8 were defined in discussions with Mr. Maxwell and from information obtained in a preliminary information package to industry from GSFC defining the requirements of the Thematic Mapper.

The LANDSAT D may carry a High Resolution Pointable Imager (HRPI) in place of the MSS of the D Mission. HRPI is poorly defined, but is expected to have power, command, and data requirements similar to the Thematic Mapper.

LANDSAT E is not defined, but is likely to carry a TM, HRPI and a Synthetic Aperture Radar (SAR). In addition to these three major sensors, LANDSAT E will probably also carry low data rate sensors for atmospheric sounding to determine water vapor and aerosol content. The data from these instruments would support and qualify data from the imaging sensors.

3.3.3 HEATE

The HEATE Mission was selected out of nine possible future Explorer Missions since it appears to be the best defined and had good scientific justification. However, HEATE is also poorly defined; an instrumental working group is now just being formed to define the mission payload. Information in Tables 3.9 and 3.10 is excerpted from the two most detailed proposals received by GSFC in response to Announcement AO-6. The HEATE Mission was selected as a result of discussions with Mr. Jack Holt at NASA

Table 3.7 Multi-Spectral Scanner (MSS)

PHYSICAL CHARACTERISTICS:

Size: Scanner - 36 cm x 38 cm x 10.7 meters
Multiplexer - 10 cm x 15 cm x 16.5 cm

Weight: Scanner - 47.6 kg
Multiplexer - 2.7 kg

SENSOR DESCRIPTION:

The Scanner is designed to scan a 160 km radius on earth, imaging six lines across in each of four spectral bands simultaneously. An oscillating mirror is at 45° to the scene and to a cassegrain telescope. The six lines in each spectral band are imaged by a 4 x 6 fiber optic array located at the telescope image plane.

INSTRUMENT DESCRIPTION:

The Scanner senses energy in the four spectral bands as follows:

Band #1	.5 to .6μ
Band #2	.6 to .7μ
Band #3	.7 to .8μ
Band #4	.8 to 1.1μ

Detectors in Bands 1 through 3 are photomultiple tubes while Band 4 detectors are silicon photodiodes.

The 24 video outputs of the detectors (6 detectors for each band) are sampled 100,000 times/sec by the multiplexer during the forward trace of the mirror. The sampled data are PAM multiplexed and sent to an A/D converter whose output is a serial bit stream at 15 megabits/second.

POWER INTERFACE:

The MSS uses raw power from regulated -24.5 volt bus (not compatible with the MSS bus) and converts this to ±15 volts and higher voltages for the PMT's range from 117 to 270 volts and another range of 1000 to 2300 volts. The inverter frequency is synchronized to the scan rate to reduce effects of inverter transients in the image data.

Table 3.7 Multi-Spectral Scanner (MSS)
(continued)

COMMAND INTERFACE:

The flight subsystem receives the following type of command pulse from the spacecraft command subsystem:

1.	Command pulse amplitude	-23.5 ± 1.0 v
2.	Command pulse width	40 ± 5 ms
3.	Maximum load current	200 ma
4.	Source impedance	60 ± 10 ohms

The MSS flight subsystem has its own internal command submatrix located in the scanner unit, but it also receives some commands directly from the spacecraft command matrix. The internal submatrix consists of a 6 MA x 7 MB real time matrix and a 4 MA x 4 MB stored command matrix.

TELEMETRY INTERFACE:

The MSS provides the following outputs to the spacecraft telemetry subsystem:

1. Analog

Range:	0 to -6.374 VDC
Output Impedance:	10 Kilohms maximum
Effective accuracy:	8 bits
Resolution:	25 mv

2. Digital (single bit words)

Off condition:	-0.5 ± 0.5 VDC
On condition:	-7.5 ∓ 2.5 VDC
Output impedance on:	1 megohm maximum
Output impedance off:	50 kilohms maximum

The MSS does not have the capability to store telemetry data. The telemetry cycling period is 16 seconds.

Table 3.8 Thematic Mapper (TM)

PHYSICAL CHARACTERISTICS:

Maximum Weight - 295 kg
Size/Configuration - Undefined

SENSOR DESCRIPTION:

The Thematic Mapper is a six-band, multispectral high resolution scanner capable of fulfilling the observational requirements of the Earth Observation Satellite program, i.e., improved land use, water resources and food supply/distribution management. The instrument consists of primary imaging optics, scanning mechanism, spectral band discrimination optics, detector arrays, radiative cooler, in flight calibrator, and required operating and processing electronics. It will collect, filter and detect radiation from Earth in a swath 185 Km track scan while the orbital progress of the spacecraft provides scan along the track. Several lines are scanned simultaneously to permit suitable dwell time for each resolution element. The variation in radiant flux passing through the field stop onto the photo and thermal detectors create an electrical output which represents the radiant history of the line. The instrument will also be capable of quantizing and multiplexing signals from all its data channels into a single digital data output stream for transmission to the ground.

Spectral Band (Micrometers)	Ground Resolution (Meters)
0.45 - 0.52	30
0.52 - 0.60	30
0.63 - 0.69	30
0.76 - 0.90	30
1.55 - 1.75	30
10.40 - 12.50	120

INSTRUMENT DESCRIPTION:

Instrument design is not well established at present. State-of-art silicon arrays and preamplifiers will be used in channels 1-4. Detectors for channels 5 and 6 are undefined. All signal processing of detector signals is to provide a wide band serial output digital data stream that includes housekeeping and has a maximum bandwidth of 120 megabits/second.

Table 3.8 Thematic Mapper (TM)
(continued)

POWER INTERFACE:

Maximum power is 400 watts, including thermal control power, if needed. Regulation and conditioning of +28V. Power from MMS to be provided by the thematic mapper. Special requirement to synchronize all converters to scan rate or multiples thereof to minimize random interference in data.

COMMAND INTERFACE:

Twenty-one discrete commands have been identified at present. Electrical characteristics of command interface are not defined.

TELEMETRY INTERFACE:

Wide band data link not to exceed 120 megabits/second will contain all imaging data and sensor calibration data. Housekeeping and Engineering data will be transmitted via spacecraft telemetry system.

Typical Housekeeping data is as follows:

First stage cooler temperature
Second stage cooler temperature
All radiometer housing temperature sensors
Scan Drive Current
Band 5 detector bias
Band 6 detector bias
Calibration Source(s) current(s) and/or power
Temperature points
Heater power status
Verification of Key Command Events

Table 3.9 Transient Gamma Ray Explorer (Experiment proposal by Los Alamos Scientific Laboratory)

PHYSICAL CHARACTERISTICS:

Weight: 190 kg (complete instrument package) No size or configuration data on complete package.

SENSOR AND INSTRUMENTATION DESCRIPTION:

1. Array of six NaI scintillators with 512 channel pulse height analyzer providing 1 millisec resolution.

2. Gamma-ray vectrometer - external photomultiplier tube viewing a CsI cube. Cube is 18 cm on a side with five 1.25 cm diameter NaI crystals on inner surface of CsI cube.

3. 14 plastic scintillators with a photomultiplier tube viewing each.

4. 2 - charged particle detectors.

5. Tri-axial magnetomoter.

6. Cobalt 60 calibration source.

COMMAND INTERFACE:

65 discrete on/off commands are required.

POWER AND TELEMETRY INTERFACE:

Instrument	Total Power	Date Rate
NaI Scintillators	8.0 w	600 b/sec.
Gamma-ray vectrometer	5.0	250 b/sec.
Plastic scintillators	6.0	450 b/sec.
Charged particle detectors	1.0	(TBD)

Table 3.10 Temporal X-Ray Explorer (Experiment proposal by GSFC/ Cambridge Astrophysics)

SENSOR DESCRIPTIONS:

1. Sky Survey camera. Comprises 4 "pinhole" cameras. Each camera is a square pyramid - 40 cm x 40 cm base, 30 cm high. A 36 x 36 element array is within base and pin-hole is at apex. The cameras are mutually oriented as diagonals to a cube and together, cover the zenith hemisphere simulataneously.

2. Large Area collimated proportional counter array is mounted on 2-axis gimbal to point anywhere within 113° of the local zenith. Provides high-resolution study of X-ray sources.

POWER INTERFACE:

Experiment power required is 30 watts.

COMMAND INTERFACE:

Sky Survey Camera: 16 discrete commands. Large Area Array: 32 discrete commands. 9-bit magnitude command for coordinate transformation.

DATA/TELEMETRY:

Sky Survey Camera: 3200 bits/sec. (400-8 bit words - time resolution 10 sec.). Large Area Array: 2544 bits/sec. (240 - 8 bit words on TM minor frame, 78-8 bit words on subframe, time resolution 1 μsec.).

Headquarters who reviewed all of the future Explorer Missions with ECON personnel. Mr. Marius Weinreb discussed details of the sensors and recommended excerpting from the two proposals.

3.3.4 GRE

The Gamma Ray Explorer is included for this study although it is no longer considered an ongoing program. It was cancelled due to cost considerations. However, it had excellent scientific justification and a detailed program plan and was thus considered valuable to the SIM investigation. Details on GRE were obtained through discussions with Mr. Frank Cepalino of GSFC and the Project Plan for the Energetic Gamma Ray Explorer Telescope (EGRET), dated November 1975. Table 3.11 represents sensor details.

3.3.5 TIROS N and O

Since the TIROS O meteorological satellite is largely undefined, it was decided to use the TIROS N Sensor complement as a model for both TIROS N and TIROS O Missions. The major sensors on TIROS N are the Advanced Very High Resolution Radiometer (AVHRR), the TIROS Operational Vertical Sounder (TOVS), and the Space Environment Monitor (SEM). Details of these sensors are presented in Table 3.12, 3.13 and 3.14, respectively. Information regarding TIROS N sensors was obtained informally from the RCA Astroelectronics Division.

3.3.6 SEASAT-A

The SEASAT-A carries five major sensors, which are defined in Table 3.15. Information on these sensors was obtained through correspondence between Mr. B. P. Miller of ECON and the Jet Propulsion Laboratory in Pasadena, California.

3.4 Methodology for Definition of Future Sensor Interface Characteristics

Because of very limited technical difinition on many sensors, a method was established for relating these sensors to existing sensors having detailed technical documentation and established

Table 3.11 Energetic Gamma Ray Explorer Telescope (EGRET)

PHYSICAL DESCRIPTION:

The EGRET comprises an Anti-coincidence dome, a Spark Chamber telescope, a total absorption shower counter and a bulkhead pedestal. The sensor payload is roughly cylindrical of 1.64 diameter and 2.25 meters long. Payload weight is 1270 kg.

SENSOR DESCRIPTION:

The major functions of the sensor are to identify high energy gamma ray events and determine energy and direction of the incident gamma ray. The sensor functions by producing positron-negatron pairs in a tantalum plate. Trajectories are analyzed in a spark chamber and energy determined in a total absorption shower counter. The anti-coincidence dome negates readings from the cosmic ray background and a time-of-flight analysis rejects gamma ray events that enter the telescope from the opposite direction.

INSTRUMENTATION DESCRIPTION:

The Sensor electronics include high voltage power supplies for PMT's high voltage pulsers (3 Kv) for the spark chambers summing amplifiers, pulse height analyzers, high discrimination timing circuits, and digital logic.

POWER INTERFACE:

Power for the experiment is 28 ± 7 volts with a negative ground. Converters within the sensor provide high voltage (3 Kv) and low voltage for analog detection and processing circuitry and digital logic. Total power required is 80 watts.

COMMAND INTERFACE:

Two distinct types of commands are utilized in the experiments: Relay power commands: At least 38 power switches are required to control the various subsystems. Data Stream Commands: About 100 bits of logic bi-level commands are required to control experiment status. These are serial data words from the spacecraft command system which would be stored in execution registers in each of the experiment subsystems.

Table 3.11 Energetic Gamma Ray Explorer Telescope (EGRET) (continued)

TELEMETRY INTERFACE:

Telemetry data is in both digital and analog form. The three major types of digital housekeeping data transmitted are:

Count rate data: All counter and coincidence rates are monitored. Rates from counters performing similar functions are commutated into the same telemetry word. The 49 pair coincidence rates are also commutated.

Status Data: The states of all command bits and of all internal switches are monitored and transmitted periodically at a low rate.

Live Time: A counter is provided to monitor the fraction of time the detector can accept gamma ray events.

Analog sensors monitor the critical functions such as experiment temperatures, power drains, operating voltages, and gas pressure. Signal conditioning power must be supplied by the spacecraft so that experiment condition may be monitored even with main experiment power off.

Table 3.12 Advanced Very High Resolution Radiometer (AVHRR)

PHYSICAL DESCRIPTION:

Visible and Infrared imaging scanner utilizing a 20 cm cassegrain telescope.

Size: 77 cm by 36 cm by 25 cm
Weight: 27 kg

SENSOR DESCRIPTION:

Sensor provides visible and thermal maps of the earth. Scanning is provided across the orbital path in four spectral bands: .55 to .9, .725 to 1.0, 3.55 to 3.93 and 10.5 to 11.5 micrometers. Ground resolution is 1.14 km.

INSTRUMENT DESCRIPTION:

The Sensor uses solid state detectors for all channels. IR channels are passively cooled to 105°K. A/ϕ converter provides a serial digital data output.

POWER INTERFACE:

Primary 28 v. power is converted for analog and digital circuitry. No high voltages are required. Power consumption is 27 watts.

COMMAND INTERFACE:

28 discrete commands are required

TELEMETRY INTERFACE:

Analog and Digital telemetry is required. There are 20 analog parameters and 14 digital discrete functions. Data output is 40 kilosamples per channel or a total digital rate of 1.6 megabits/second.

Table 3.13 TIROS Operational Vertical Sounder

Unit	Description	Size (cm)	Weight (kg)	Power (Watts)	Commands	Telemetry		Data Rate
						Analog	Digital	
Basic Sounding Unit	Temperature and Humidity profiling from earth to stratosphere	36 x 64 x 36	28.5	25.2	15	13	9	2880 bits/second
Strato-spheric Sounding Unit	Stratospheric Soundings - Altitude 25-50 Km	38 x 26 x 37	18.1	18.6	5	10	4	480 bits/second
Micro-wave Sounding Unit	Lower Atmosphere soundings - 0 to 20 Km independent of cloud cover	two units 66 x 50 x 36 25 x 20 x 16	26.2	32.4	15	10	8	320 bits/second

Table 3.14 Space Environment Monitor (SEM)

PHYSICAL DESCRIPTION:

The SEM Comprises three sensor units and a Data Processing unit. Size and weight for each are as follows:

Unit	Size (cm)	Weight (kg)
Total Energy Detector (TED)	36.1 x 14.5 x 13	2.7
Medium Energy Proton & Electron Detector (MEPED)	21 x 11.2 x 20	3.1
High Energy Proton & Alpha Detecotr (HEPAD)	16.8 x 9.7 x 26.9	2.7
Data Processing Unit	30.5 x 28.5 x 6.9	3.1

SENSOR DESCRIPTION:

The SEM enables determination of energy from solar particles in the upper atmosphere. Employed as a solar event warning system.

INSTRUMENT DESCRIPTION:

TED: Deflection Analyzer and Channeltron
MEPED: Solid State detector telescope and omnidirectional detectors
HEPAD: Ceverkow Scintillator and photomultiplier sensors.

POWER INTERFACE:

Power is converted from +28 volts to low voltages required by analog and digital circuits. The HEPED requires high voltage (1 to 3 kilovolts) for the photomultiplier tubes. Total power required: 11.3 watts input.

COMMAND AND TELEMETRY INTERFACE:

The DPM acts as an interface unit for command and telemetry channels for the three sensors. Requirements are as follows: Commands: 12 discrete on-off functions. Telemetry: Digital Discrete: 15. Analog: 15. Serial Digital Data: 160 bits/second.

Table 3.15 Sensors for SEASAT-A

Sensor	Description	Size (m)	Weight (kg)	Power (Watts)	Scientific and Engineering Data Rate
SMMR (Microwave Radiometer)	6.6 to 37 GHz. Measures ocean surface temperature and wind speed, integrated moisture and rain droplet size	(not available)	47	61	2 K bits/seconds
SCAT (radar scatterometer)	14.6 GHz. Measures ocean wind speed and direction	Antenna: 2.69 long Electronics: 1.0 x 4 x .35	Antenna: 13.6, Electronics: 80	80 - regulated 85 - unregulated	540 bits/second
VIRR (Visible and infrared radiometer)	Visible and IR imagery - resolution to 2 Km	(not available)	9	8 (-24.5 volts unregulated)	1800 Hz analog bandwidth
SAR (Synthetic Aperture Radar)	Radar imagery and wave spectra	1.0 x .8 x .25	100	401 to 558 - RF power level and PRF dependent	Engineering data 500 bits/second 19 MHz analog data link
ALT (Radar Altimeter)	Sea height - accuracy ± 10 cm (1)	1.2 dia. x .2 h.	70	150	8.15 K bits/second

cost histories. The method relates poorly defined sensors to known sensors in terms of the three interfaces of concern and associated electronics that may be affected or replaced by a Standard Interface Module. A specific mission sensor might be related to known "sensor A" for its power interface while it might be related to a different "sensor B" for the data interface. By using this method, the selected mission sensors can be compared with a relatively few known sensors.

For the selected missions, the sensors of TIROS-N and SEASAT-A are well defined as is the Multi-spectral Scanner (MSS) of LANDSAT.

The following is the rationale for the comparisons made in Table 3.16.

3.4.1 AASIR

The AASIR is compared with the MSS primarily because of scanning mechanism and output data rate similarity. The primary difference between the two sensors is the additional IR sounding channel of the AASIR which is not truly represented in the MSS.

3.4.2 MASR

The MASR and SEASAT Radiometer (SMMR) comparison is quite good, in terms of the sensor type, number of channels, data processing, and overall power consumption. Major differences between the two sensors are that the MASR has a much larger antenna which must be programmed to scan and point in specific directions, whereas the SMMR has a fixed scan format. Power, data, and command interfaces should be similar, with the exception of the requirement for magnitude commands for the MASR.

3.4.3 Thematic Mapper and HRPI

Both the Thematic Mapper and HRPI are compared with the LANDSAT MSS. The MSS is the most advanced imaging scanner that has a good cost history. However, both the Thematic Mapper and

Table 3.16 Sensor Comparison Matrix

Mission / Sensor — Cost Reference Sensor	STORMSAT		LANDSAT D & E			HEATE						GRE		
	AASIR	MASR	TM	HRPI	SAR	NaI Scintillator	γ-Ray Vectrometer	Plastic Scintillator	Charged Particle Detector	Sky Survey Camera	Large Area Proportional Telescope	Anti-coincidence Dome	Spark Chamber Telescope	Total Absorption Shower Counter
SEASAT														
SMMR (Microwave Radiometer)		P, D												
SAR (Synthetic Aperture Radar)					P, D									
TIROS-N														
HEPAD (High Energy Proton/Alpha Detector)						P	P	P		P	P	P		P
MEPED (Medium Energy Proton/electron detector)									P					
DATA PROC. UNIT						D	D	D	D	D	D	D	D	D
LANDSAT														
MSS SCANNER	P		P	P										
MSS MULTIPLEXER	D		D	D										
SAS-2													P	

NOTE: P = Power; D = Data

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the HRPI represent significant advances in technology and complexity over the MSS. Data output rates of the Thematic Mapper and HRPI are in the order of 100 megabits/second, making data handling comparisons somewhat uncertain. Power conditioning for both the Thematic Mapper and HRPI should be similar to MSS, but it is not expected that there will be requirement for the high voltage power supplies on MSS which uses PMTs in the visible channels.

3.4.4 Synthetic Aperture Radar

The SAR for LANDSAT is compared with the SAR for SEASAT. Significant increases in complexity in the LANDSAT SAR are expected as this unit will utilize two frequencies, two polarizations, and will have a significantly wider information bandwidth and hence, higher power consumption than the SEASAT unit.

3.4.5 HEATE and GRE Instrumentation

The instruments on these explorer missions are compared with the TIROS-N Space Environment Monitor Sensors. The HEPAD on TIROS-N uses scintillators, photomultipliers, pulse height analizers, which are similar to several of the experiments on the Explorers.

The type of circuitry for power and data interfaces is similar. The only requirement is to properly adjust complexity factors between the Explorers and the SEM. The only real deviation power interface is the case of the spark chamber telescope for the GRE which requires high voltage pulsers which, at the time of a gamma-ray event, activate the telescope.

3.4.6 Other Missions and Sensors

Using the methodology described above, interface characteristics of additional sensors from MSS missions in the May 1976 model have been tentatively defined. The matrix in Table 3.17 shows the comparison to known sensors.

Table 3.17 Mission Comparison Matrix

Cost Reference Mission \ Mission	Space Telescope	HEAO Block II	TIROS N' &O	SEASAT B & C	Technology Demonstration Satellite	Earth Survey Satellite
SEASAT-A				X (all sensors)	X (SAR only)	
TIROS-N		X (SEM only)	X (all sensors)			
LANDSAT-D	X (MSS and RBV)					X (MSS and TM)

The comparisons of Table 3.17 are more general because of rather sketchy information available regarding many of the mission sensors. The following is a brief description of the considerations made in relating mission sensors to the reference sensors.

3.4.6.1 Space Telescope

The sensors from LANDSAT D, i.e., the Multi-Spectral Scanner and the Return-beam Vidicon Camera are most representative of the types of imaging sensors that would be carried on a space telescope of the MSS class.

3.4.6.2 HEAO-Block II

The most likely sensor complement for the HEAO-Block II Missions is a scale-up of the Gamma-Ray Explorer instrumentation. Reference sensors for GRE may be used. However, HEAO-Block II, as described in the 1973 Mission Model, is a considerable scale-up from the GRE size causing significant differences in power conditioning, data handling, and command interfaces.

3.4.6.3 Technology Demonstration Satellite

TDS will carry developmental sensors and spacecraft components as a means of shortening overall development time for new technology. A synthetic aperture radar is planned for the first flight.

3.4.6.4 Earth Survey Satellite

This operational earth resources survey satellite will incorporate sensors identical to or very similar to those carried on LANDSAT D and E.

4. ANALYSIS OF STANDARD INTERFACE MODULES AND CHARACTERISTICS

4.1 Methodology for Selection of Standard Elements

A systematic approach was developed to identify the system elements which have a degree of commonality within each sensor package, within each of the sensor support systems, and from one type of mission payload to another. An outline of this approach is given below.

A. Generate conceptual drawings for each mission payload which depict basic functional flow for each sensor and all identifiable hardware.

B. Establish an integrated equipment list for each mission payload categorizing hardware elements which are 'mission peculiar', i.e., those elements which are custom made to perform a designated task for the sensor system and which would most likely have to be redesigned or altered if used on another type sensor, and those hardware elements which are 'mission common', i.e., hardware elements which can be used interchangeably from one sensor to another.

C. Characterize the mission peculiar and mission common hardware items by the various sensor or sensor support subsystems to which they belong i.e.,;

Hardware Subdivisions (typical)	Code ID
Sensor System	S
Power Conditioning	P
Data Handling	D
Science data	
Engineering data	
Inflight calibration data	I
Command Handling	C

D. Transpose the conceptual mission hardware drawings into logic flow charts by assigning specific type and kind numbers to each hardware element in accordance with the Kaman Sciences GO methodology. A brief description of the GO methodology is given in Appendix 8.4.

E. Code the logic flow charts into a set of functional GO models and determine the relative sensitivity of each kind of hardware element used in the various mission/sensor models. Coding work is considerably simplified by making use of GO 'Supertype' elements which need to be detailed only once. These elements are called out by a Supertype ID whenever the same logic element is required somewhere else in the mission sensor or supporting subsystems.

F. Exercise the GO computer runs to determine frequency of mission hardware usage, degree of hardware sensitivity, and probability of sensor success as a function of hardware changes if so desired.

G. Consider all hardware in the mission common category as potential hardware for Standard Interface Modules (SIM's). Examine the potential hardware in each hardware subdivision giving priority to high usage and hardware combinations that will minimize sensitivity. Define an initial set of SIM's.

H. Define preliminary technical and programmatic requirements for proposed SIM's to serve as primary input for the costing evaluation phases.

The following section will be used to discuss the application of paragraphs (A) through (F) in the above approach to the selected missions. Section 4.3 will discuss the results of paragraph G, while the programmatic and technical requirements (paragraph H) will be summarized in Section 4.4.

4.2 Conceptual Mission Payloads

All Functional Block Diagrams and GO Logic Diagrams referred to in this section are found in Appendix 8.5.

4.2.1 STORMSAT

Conceptual block diagrams of the STORMSAT payload are shown on drawings A1-A3. These drawings depict the essential components and functional flow constituting the two sensors used in STORMSAT. The STORMSAT sensors are identified as follows:

ADVANCED ATMOSPHERIC SOUNDER AND IMAGING RADIOMETER (AASIR)

MICROWAVE ATMOSPHERIC SOUNDING RADIOMETER (MASR)

AASIR:

The GO logic model assumed that the calibration functions, the step/scan functions, any filter wheel action, and the IR or visible focusing functions are all essential for proper operation of the AASIR. In addition, it was assumed that an automatic focus adjustment is brought about in the visible or IR wave trains whenever two out of three thermocouples (Signals 207-209, or signals 233-235) indicate an unbalance above threshold in the temperature-focus control servo loops. The GO logic flow starts with the admission of visible or IR radiation from earth or the intervening atmosphere (Signal 183), which is passed to the primary and secondary mirrors of the folded Cassegrain system (Elements 1-108 & 1-109) and through a calibration shutter. At the end of each step-scan cycle, a calibration pulse signal is generated. This signal will actuate the calibration motor, shutter, and the calibration ramp signal generator (Signals 186-190) to image blackbody calibration signals into the optical path to the detectors. The AASIR is a fast step-scan device. It was assumed that a scan mirror is used for the fast N-S scan and the whole gimbal is torqued for the slower E-W stepping action. Step and scan 'SYNC' signals (Signals 176 & 220) are provided from the STORMSAT Data Sequencer and Controller. Since this represents a feedback, the 'SYNC' signals are introduced as type 5 inputs. After passing through the scan mirror the optical signal is split off into a visible imaging wave train and an IR wave train. The visible light train is focused on two sets of detector arrays, one with nine detectors for IR Imaging (Supertype 102, Signals 264-272), and the other, an array of 18 sounding detectors (Supertype 103, signals 273-290). The outputs of these 48 detectors and appropriate step-scan resolver signals (signals 224 & 341) are fed to a common Data Handling Unit along with possible signals from the MASR sensor.

MASR:

The signal flow path through the passive MASR sensor is very similar to that through the AASIR device. Energy in the microwave region is sensed by the dual microwave antenna horns (type 1-121), which pass the composite signal (signal 308) to an array of 16 mixers, multiplexed filters, IF amplifiers, and 2nd detectors (Supertypes 108 & 109). The antenna reflector is step-scanned in synchronism with the AASIR telescope system. The 2nd detector output (signals 215-338) and resolver signals from the MASR are fed directly to the telemetry RIU, or to the common Data Handling Unit previously mentioned. The operating frequencies for the MASR are selected to be close to the H_2O & O_2 atmosphere absorption bands to provide higher sensitivity and to mask out a number of competing emissions.

DATA HANDLING:

STORMSAT will carry an operational Data Handling Unit (DHU) and a standby DHU which can be commanded 'on' if the operational unit should fail (Supertypes 110) The DHU is used to multiplex the analog outputs from the detectors in both STORMSAT sensors, convert to 8 & 10 bit digital words and buffer for subsequent transmission and recording (signals 370-383) on a special data link. The sensor engineering and housekeeping data, assumed to be less than 10 kbps, and possibly the MASR data are passed directly to the spacecraft (S/C) telemetry RIU for digitizing and formating on the S/C Telemetry system, (signals 379-384). Data rates for the AASIR are estimated to be 10.5 Mbps and for the MASR about 1 Kbps. Since the STORMSAT vehicle is geosynchronous it was assumed possible to lower the data rate on the AASIR to less than 1 Mbps if a standard data link is desired. Command data rates for either sensor are assumed to be less than 2 Kbps and will be handled by the S/C command RIU (signals 170-175).

POWER CONDITIONING:

Since the STORMSAT payload was in a conceptual stage at the time of this study, the power conditioning for STORMSAT had not yet been defined. It was assumed as a first alternative that the power conditioning could be similar to the power conditioning subsystem to be proposed for the LANDSAT mission. Both STORMSAT & LANDSAT require Logic Power Supplies (LPS) for digital circuitry, Regulated Power Supplies (RPS) for LSI analog circuitry, and Pulsed or Unregulated power (PPS/UPS) for heaters, step motors, scan-motors, etc. To allow a comparison of sensitivity data with power systems on other missions, the Regulated Power Supply for STORMSAT was deliberately modeled with single power units instead of dual units. The Logic power and the Pulsed or Unregulated Power Supplies were assumed to be dual units incorporating internal redundancy.

SENSITIVITY & USAGE:

Table 4.1 itemizes the results of the functional sensitivity evaluations on STORMSAT by various hardware categories and

Table 4.1 STORMSAT Sensitivity Results

		Kind	Sensitivity	Usage
Mission Common				
P	EMI Filter	101	2	2
	Line Filter	102	1	1
	Transformer (Prim & Sec.)	103	1	1
	Transformer Taps	104	47.8	56
	Clock Regulator	105	1	1
	Inverter Feed Line (output)	126	47.8	60
	Rectifier	127	31.9	34
	Voltage Regulator	128	31.9	34
	Power Filter	129	31.9	36
	Clock	601	0	2
	Inverter	637	0	2
D	Clock Regulator	105	1	1
	Sync Code Generator	130	0	2
	S/C Telemetry	131	0	1
	Clock	601	0	2
	RIU	602	2	2
	S Band Transformer	629	1	1
	S/C Tape Recorder	630	0	1
	Analog Multiplier	640	0	6
	Sample & Hold Circuits	641	0	6
	ADC	642	0	6
C	Common Receiver	106	1	1
	Central Computer	107	1	1
	RIU	602	2	2
I	Calib. Ramp. Gen.	603	1	1
	Calib. Driver	604	1	1
	Calib. Motor	606	1	1
	Calib. Mirror	623	1	1
S	Step/Scan Mirror	115	1	1
	Detec Temp Monitors	119	0	6
	Radiation Cooler	120	1	1
	Step Driver	610	1	1
	Position Resolver	612	4	4
	Focus Drive	614	2	2
	Scan Drive	615	1	1
	Filter Wheel Mtr	620	1	1
	Filter Wheel Drive	621	1	1
	Scan Motor	617	1	2
	Chopper Drive Circuit	622	1	1
	Systems Heater	658	1	1
	Second Detectors	627	16	16

Table 4.1 STORMSAT Sensitivity Results (continued)

		Kind	Sensitivity	Usage
Mission Peculiar				
D	Digital Formatter	643	0	2
	Oscill. Sequencer	644	0	2
	Output Buffer	645	0	2
I	Calib. Shutter	607	1	1
S	Primary Optics	108	1	1
	Secondary Optics	109	1	1
	Visible Imaging Prism	110	1	1
	IR Folding Mirror	111	1	1
	Fixed Folding Mirror	112	1	1
	Visible Collimating Lens	113	1	1
	IR Collimating Lens	114	1	1
	IR Field Lens	116	1	1
	Relay Lens Assembly	117	2	2
	Sensor Window Assembly	118	1	1
	Feed Horns (microwave)	121	1	1
	Antenna Gimbals	122	2	2
	Diplexer & Selected Filters	123	6	6
	Multiplexer & Selected Filters	124	7	7
	Polarization Splitter (OMT)	125	2	2
	Step Torquer	608	2	2
	Filter Wheel	619	1	1
	Logic Timer	625	1	1
	Local Oscillator	626	2	2
	Solid State Photo Detectors	631	48	48
	Mixer	638	4	4
	IF Amplifiers	639	8	8
	Chopper	646	1	1
	Temp Control Servo	613	2	2

NOTE: Kind numbers refer to component number on GO diagrams

P = Power Conditioning Hardware
D = Data Handling Hardware
C = Command Handling Hardware
I = In-flight Calibration Hardware
S = Sensor Hardware

also indicates the 'kind' number assigned to a given kind of component on the GO logic diagrams and a usage number which defines the number of times a given component is used in the onboard sensors for any given flight of that mission.

Sensitivity is defined as the partial derivative of system reliability to component reliability, i.e.; if a given kind of component has an actual reliability during operation which is Δc different from the initially assumed value (all components in this study were assumed initially to be perfect) the degradation in total system reliability is equal to Δc x the sensitivity value given in the last column of the sensitivity tables. It is obvious that improved system performance can be obtained by maintaining the functional sensitivity as low as possible.

4.2.2 LANDSAT

Conceptual block diagrams of the LANDSAT payload are shown on drawings B1-B3. These drawings depict the essential components and functional flow that make up the two sensors planned for LANDSAT, i.e.,

MULTI-SPECTRAL SCANNER (MSS)

THEMATIC MAPPER SYSTEM (TMS)

These drawings should be compared to the equivalent GO Logic diagrams found in drawings B4 and B5.

MSS:

The GO logic diagram for the MSS is initiated with Signal 221 representing visible and IR energy entering the optical aperture. This signal is passed through a folded set of primary system optics (kinds 1-108 and 1-109), through a calibration shutter and is focused on the end of an optical fibre bundle. The scanning mirror is driven by a scan motor and driver (Signals 222-225). Calibration is performed using deep space, sun images

(Signal 227-228) and a calibrated lamp source (Signal 244-251). The optical fibers pass the signal along to an array of 18 photomultiplier tubes covering the IR bands (Supertypes 112 and 113). Scan position is obtained from a set of solid state monitoring detectors, which determine the direction of scan by using signal slope comparators (Signal 230-241). The output of the detectors is fed to a pair of common Data Handling Units (Supertype 116).

TMS:

The TMS is very similar in principle to the MSS, only more advanced. Information coming late into this study indicates that the TMS utilizes almost 100 detectors compared to 30 on the MSS. It was initially assumed during the construction of the GO model that the TMS had 56 detectors instead of 100. The ground resolution for TMS is about 25 meters compared to 75 or 80 meters on the MSS. The GO diagram assumes auto-focusing on both the visible and IR bands, filter selection, electronic reticules, black body calibration, and a scan mirror drive with appropriate position resolver outputs. The visible and IR wave trains feed into the detectors (Supertypes 114 & 115), and the detector outputs are passed to a common Data Handling Unit (Supertype 116).

DATA HANDLING:

It was assumed that LANDSAT will carry a standby compliment of DHU's which can be commanded 'on' if the operational units should fail. The DHU's for LANDSAT must be able to handle a total of 130 Mbps, 115 Mbps from the TMS, and 15 Mbps from the MSS. Initial plans call for 8 bit A/D conversion. Present state of the art in DHU's would require about 24 local DHU's for the TMS, (four for each band) and possibly 3 local DHU's for the MSS. Due to the fast data rate, each of these DHU's may require about four parallel Sample and Hold circuits and four ADC circuits. Both the TMS and the MSS will require one or two ultra-fast Master

Digital Multiplexers to combine these dense data streams into a special data link. For sensitivity evaluations only 2 DHU's were assumed to be on board the LANDSAT. The sensor engineering and housekeeping data for LANDSAT, assumed less than 10 Kpbs, is passed directly to the spacecraft telemetry RIU for digitizing and formating on the S/C telemetry system. Command data rates for the two sensors on LANDSAT are assumed to be less than 2 Kbps.

POWER CONDITIONING:

Power conditioning for the MSS mission is well defined, but the power conditioning for the TMS is not defined as yet. For purposes of this study it was assumed that the type of power conditioning used for MSS would also apply for the TMS. To allow sensitivity comparisons it was assumed that the Regulated Power Supplies and the High Voltage Power Supplies were not redundant units. All other power supplies were assumed to be redundant.

Sensitivity & Usage:

Table 4.2 itemizes the results of the functional sensitory evaluations on LANDSAT by various hardware catagories and also indicates the 'kind' number assigned to a given kind of component on the GO logic diagrams and a usage number which defines the number of times a given component is used in the onboard sensors for any given flight of that mission.

4.3.2 TIROS

Sensors for TIROS R&D flights prior to 1980 are well defined. Sensors for the TIROS flights during 1981-1985 were assumed to be updates of the 1980 configuration. Block diagrams of the assumed payloads are shown in drawings C1-C4. These drawings depict the essential components and functional flow constituting the assumed TIROS payload and include the following sensors:

Table 4.2 LANDSAT Sensitivity Results

		Kind	Sensitivity	Usage
Mission Common				
P	EMI Filter	101	2	2
	Line Filter	102	1	1
	Transformer (Pri & Sec.)	103	1	1
	Power Feed Lines (TAPS)	104	56.8	67
	Clock Regulator	105	1	1
	Inverter Feed Lines (output)	126	56.8	69
	Rectifier	127	37.9	40
	Voltage Regulator	128	37.9	40
	Power Filter	129	37.9	42
	Pass Network	136	3	3
	Isolation Transformer	137	3	5
	DAC Converter	138	1	2
	Clock	601	0	2
	Inverter	637	0	2
	High Volt. Multiplier	659	3	3
D	Clock Regulator	105	1	1
	Sync Code Generator	130	2	4
	S/C Telemetry	131	0	1
	Clock	601	0	2
	RIU	602	2	2
	S Band Transponder	629	1	1
	S/C Tape Recorder	630	1	1
	Analog Multiplier	640	0	6
	Sample & Hold Circuits	641	3	6
	ADC	642	3	6
C	Command Receiver	106	1	1
	Central Computer	107	1	1
	RIU	602	2	2
I	Sun Calib. Mirror	132	1	1
	Calib. Source	135	2	2
	Calib. Motor Drive	604	1	1
	Calib./Shutter Motor	606	1	1
	Calib. Lamp Drive	653	2	2
	Calib. & Shutter Drive	654	1	1

Table 4.2 LANDSAT Sensitivity Results (continued)

		Kind	Sensitivity	Usage
Mission Common				
S	Detec. Temp. Monitors	119	0	2
	Radiation Cooler	120	1	1
	Slope Comparator	649	3	3
	Position Resolvers	612	2	2
	Focus Drive	614	2	2
	Scan Drive	615	2	1
	Filter Motor	620	2	2
	Filter Drive	621	2	2
	Scan Gate (End)	651	1	1
	Sys. Heater	658	1	1
	Scan Motor	652	2	2
	SIG Compression Amp	662	1	2
Mission Peculiar				
D	Output Buffer	645	1	2
	Oscill. Sequencer	644	2	4
	Digital Formatter	643	2	4
I	Calib. Shutter	607	2	2
S	Primary Mirror	108	2	2
	Secondary Mirror	109	2	2
	Optical Fiber Bundle	133	1	1
	Scan Monitor Detec	647	1	1
	Monitor Preamp	648	1	1
	Temp. Control Servo	613	2	2
	Imaging Optics	656	2	2
	Filter Wheel	619	2	2
	Correc Optics & Elec	657	2	2
	Reticule			
	Sig Compress Amplifier	662	1	2
	Solid State Photodetector	663	68	68

NOTE: Kind numbers refer to component number on GO diagrams

P = Power Conditioning Hardware
D = Data Handling Hardware
C = Command Handling Hardware
I = In-flight Calibration Hardware
S = Sensor Hardware

ADVANCED VERY HIGH RESOLUTION RADIOMETER (AVHRR),
TIROS OPERATIONAL VERTICAL SOUNDER (TOVS), i.e.,
- BASIC SOUNDING UNIT (BSU)
- STRATOSPHERIC SOUNDING UNIT (SSU)
- MICROWAVE SOUNDING UNIT (MSU)

SPACE ENVIRONMENT MONITOR (SEM), i.e.,
- TOTAL ENERGY DETECTOR (TED)
- MEDIUM ENERGY PROTON-ELECTRON DETECTOR (MEPED)
- HIGH ENERGY PROTON-ALPHA DETECTOR (HEPAD)

DATA COLLECTION SYSTEM (DCS)

The drawings for TIROS should be compared with the equivalent GO logic diagrams C5-C7.

AVHRR:

The AVHRR is a high resolution (approximately 1 Km) imaging radiometer. The logic flow is quite similar to the AASIR and MSS sensors. Radiant energy in both the visible and IR wavelengths is admitted into the instrument aperture and falls upon the scanning mirror (Signals 201-207). The scanning mirror passes the signal through the Cassegrain optics to a beam splitter mirror that separates the visible and IR wavelengths into two trains (Signals 207-211). There are two solid-state detectors located in the visible train with appropriate post amplification (Signals 211-227), and three detectors with post amplification in the IR train (Signals 228-240). At this point, the detected signals are passed to a local Data Handling Unit, i.e., analog multiplexer, Sample and Hold, ADC, output buffer, etc. (Signals 241-256).

TOVS-BSU:

The Basic Sounding Unit (BSU) is similar to the sounding channels in the AASIR sensor. The BSU admits radiant energy to a scanning mirror and from thence to one of seven optical telescope trains mounted on a chopper wheel, (Signal 269-286). Each optical telescope (Supertype 110) consists of folded Cassegrain optics, and beam splitter optics with selective filters which form two channels from each of the seven primary channels, making a total of 14 sounding channels. Each sounding channel has a solid-state detector, appropriate amplifiers and a signal conditioner. The 14 sounding outputs are passed to a local Data Handling Unit to be converted to digital information and held in a temporary buffer register (Supertype 111), where the information is eventually accessed by the TIROS Information Processor (TIP).

TOVS-SSU:

The Stratospheric Sounding Unit (SSU) is a United Kingdom sensor which allows extremely selective monitoring of key energy bands in the upper reaches of the atmosphere. The detectors consist of three low pressure CO_2 gas cells which are modulated at preselected rate, (Supertype 100). The signals being analyzed are passed through this gas by means of an optical-filter device. A pyroelectric detector is used to monitor the transmission of the gas cell. The detector is followed by am amplifier, a phase shift detector, and an integrator to monitor the slope of the detected signal, (Supertype 101). The final output signal is passed to a local Data Handling Unit (Supertype 102), where it is digitized and passed to the TIROS Information Processor.

TOVS-MSU:

The Microwave Sounding Unit (MSU) is very similar to the MASR sensor used on STORMSAT. Microwave energy is received on each of the two scanning reflector antenna systems (Signals 201-365, 201-376). Orthomode Transducers (OMT) are used to split the energy into four channels, i.e., two polarizations for each of the two input channels. Each of these four channels are fed into a Dicke type receiver, where the incoming signal is compared at one KHz rate with a reference load. This modulated signal is subsequently passed to a local mixer, IF, and detector stage, followed by a video amplifier, phase detector, and dc amplifier (Supertype 130). The output signals are again passed to a local Data Handling Unit (Supertype 131) for conversion to digital form prior to access by the TIROS Information Processor.

SEM-TED:

The Space Environmental Monitor has been incorporated to study the spacial-temporal flux of space particles, determine the incident energy spectrum and apparent sources of these energy streams. The Total Energy Detector (TED) is one of the first of these instruments and will provide total energy measurements for both proton and electron fluxes. The TED uses a programmed swept, electrostatic, curved plate analyzer at the front end to select and separate particle type and energy, a channeltron detector (Supertypes 140 and 142), and a subsequent signal analyzer to sense and quantify the intensity of the selected energy bands, (Signals 600-617). The output of the TED is sent to a local SEM mux (Signal 668-670) prior to bring accessed by the TIROS Information Processor (TIP). The particles of interest run from 300 eV up to 20 KeV.

SEM-MEPED:

The Medium Energy Proton and Electron Detector senses protons, electrons and charged ions with energies from 30 KeV to several tens of MeV. The MEPED consists of four directional, solid-state detector telescopes and one generally omnidirectional sensor. Two of the telescopes are set up for proton or positive ion detection (Signals 618-627 and 750-755), and the remaining two, for electron or negative ion detection, (Signals 628-637). The output of each telescope is fed into a special signal analyzer which sorts each event into appropriate 'bins' which are characteristic of particle type and energy band. The outputs of each 'bin' are pulses of fixed amplitude and are fed to the SEM multiplexer, (Signals 648-649). Each of the proton telescopes have two solid-state detectors while the electron telescopes have only one. The omni-detector consists of three lithium drifted solid-state detectors mounted under spherical shell caps with carefully controlled nuclear stopping power. The outputs from these detectors are connected via charge sensitive preamps to the same signal analyzer discussed above for the telescopes. (Signals 638-647).

SEM-HEPAD:

The High Energy Proton and Alpha Detector senses protons and alpha particles from a few hundred MeV up through relativistic energies above 850 MeV. Two solid-state detectors are used to define the field of view, (Signals 651-657). The basic detector consists of a Cerenkov crystal and photomultiplier tube, (Signals 650-664). The output of all 3 detectors is passed to a special signal analyzer (Signals 661-667) which separates the incident protons into one of four energy bands and the alpha particles into one of two energy bands. The signal analyzer results are passed on to the local SEM multiplexer.

DCS:

The Data Collection System for TIROS is an onboard automatic relay system which picks up data from a large number of surface instrumentation platforms on earth, formats the data, and retransmits it to local users via the TIROS VHF beacon. The total number of instrument platforms which can be accomodated number about 2000. As many as 200 platforms, each telemetering up to four channels of data, can be located in a single 5 degree visibility circle. This instrument has not been detailed on the GO logic charts. However, it will require local power, data handling, and commands. The output of this device is fed as an input into the TIROS Information Processor (TIP) which serves as a digital multiplexer and sequencer, (Signals 673-674) for all sensors in the TVOS and SEM unit inventories.

DATA HANDLING:

As noted in the previous paragraphs on TIROS sensors, there are several levels of data multiplexing in the TIROS system, i.e., the AVHRR, each of the three sensors in the TOVS, the SEM, and the DCS each have their own multiplexing and digital conversion units. The output of the AVHRR Data Handling Unit which is at a higher rate then the others is passed directly to the TIROS Manipulated Information Rate Processor (MIRP) for subsequent recording or transmission over the real-time High Resolution Picture Transmission link (HRPT), or over the lower resolution, Automatic Picture Transmission link (APT). The DHU's for all other TIROS sensors are combined into one data stream by the TIROS Information Processor (TIP) and passed over the spacecraft VHF link to GSFC or, on command, the spacecraft recorders for delayed transmission or real time transmission over the APT and HRPT links by using the MIRP. (Signals 673-685). Housekeeping and Engineering data is shown on the GO charts (Signals 686-689) as feeding a telemetry RIU and being passed directly down the S-band link. Present TIROS flights combine the Housekeeping and Engineering data into the TIROS Information Processor and transmit it over the VHF link. TIROS missions which will fly during the time period embraced by this study (1980-1985) will use the S-band link rather than the VHF link. Command data rates for the TIROS system are assumed to be less than 2 Kbps and will be handled by the S/C command RIU, (Signals 99-140). Table 4.3 lists the data rates which must be handled by the various DHU's in the TIROS system.

Table 4.3 TIROS Data Rates		
Sensor	Rate	Link
AVHRR	0.6654 Mbps	HRPT, APT
BSU	2.880 Kbps	HRPT, APT, S-Band
BSU	.480 "	HRPT, APT, S-Band
MSU	.320 "	HRPT, APT, S-Band
SEM	.160 "	HRPT, APT, S-Band
DCS	.720 "	HRPT, APT, S-Band
Eng.	.500 "	HRPT, APT, S-Band

POWER CONDITIONING:

The TIROS system appears to have dedicated power supplies for each of the three TOVS sensors. These dedicated supplies however require 28V unregulated S/C power and S/C 5 vdc interface power. The remainder of the TIROS sensors require, in addition to the unregulated power bus (UPS), a pulsed power bus (PSS), regulated power for analog circuitry (RPS), regulated power for digital circuitry (LPS) and programmable high voltage power supplies (VHS) for the SEM sensors. the TIROS power supply modules, for this study, are conceptually depicted in 'GO' Drawing C6. To more adequately evaluate sensitivity, only three of these power modules were made redundant, i.e., the pulsed (PPS) and unregulated (UPS) power supply modules and the high voltage power supply module (VHS).

SENSITIVITY & USAGE:

Table 4.4 itemizes the results of the functional sensitivity evaluations on TIROS by various hardware categories and also indicates the 'kind' number assigned to a given kind of component on the GO logic diagrams and a usage number which defines the number of times a given component is used in the onboard sensors for any given flight of that mission.

Table 4.4 TIROS Sensitivity Results		Kind	Sensitivity	Usage
Mission Common				
P	EMI Filter	101	2	2
	Line Filter	102	2	2
	Clock Regulator	105	1	1
	Power Lines	126	47	133
	Voltage Regulator	128	29	33
	Power Filter	129	29	33
	DAC	138	0	8
	DC/DC Converter	140	0	25
	DIG IF Unit	141	0	8
	H.V. Multiplier	142	0	8
	H.V. Reg. & Filter	143	0	8
	Power Reg. & Conditioner	153	0	2
	Relay Driver	160	1	1
	Reg. & Filter	161	4	5
	Command Relay	176	1	1
	Clock	601	0	2
	CMD Switch	662	7	21
	Relay	665	3	6
	Error Amp.	667	0	8
	Amp. Control Oscillator	668	0	8
	Interface Circuits	693	2	2
D	Clock Regulator	105	1	1
	Housekeeping Multiplexer	151	1	1
	Clock	601	0	2
	RIU	602	1	2
	Tape Recorder	630	0	1
	Analog Mux	640	7	7
	S&H	641	4	4
	ADC	642	4	4
	Timing Generator	663	1	1
	Sync Generator	664	1	1
	Data Register	678	1	1
	Gray-Binary Converter	679	1	1
	Output Buffer	680	2	2
	Auxiliary Mux	690	1	1
	Digital Mux	701	3	3
	Digital Mux Driver	704	1	1
	Analog Buffer Amp.	713	0	4
	S/C Telemetry	728	0	1
	Data Transmitter, VFH,HRPT,HPT	729	1	3

Table 4.4 TIROS Sensitivity Results (continued)

		Kind	Sensitivity	Usage
Mission Common				
C	Command Relay	176	1	1
	RIU	602	1	2
	CMD Switch	662	7	21
	Relay	665	3	6
	Command Receiver	669	1	1
	Central Computer	670	1	1
I	Black Body Reference	152	7	11
	Referenced Load	165	4	4
	Shutter Solenoid	172	1	1
	Cal. Ramp. Generator	603	1	1
	Ramp Generator	666	0	2
	Voltage Reference Circuit	720	2	2
S	Radiation Cooler	120	1	1
	Beam Splitter	145	1	1
	Temp. Monitor ckt	149	2	16
	Step Motor	158	3	3
	Scan Motor	617	1	1
	Mirror Position Switch	159	1	1
	Position Resolvers	612	1	2
	Scan Mirror	652	3	3
	System Heater	658	16	16
	Post Amplifier	672	22	22
	Chopper Motor	681	1	1
	Beam Splitter Optics	685	7	7
	Temp. Control ckt	691	1	2
	Isolation Amplifier	707	4	4
	DC Amplifier	712	4	4
	Optical Position Mon.	682	1	1
	Source Gate	726	1	1
	Motor Drive	615	3	3
Mission Peculiar				
P	BSU PWR Supply RPS	154	1	1
	LPS	155	1	1
	UPS	156	1	1
	PLS	157	1	1
	MSU PWR Supply Programmer	174	1	1
	(Chan 1 through Chan 4)	175	4	4
D	Data Hold Logic	703	1	1
	Memory	738	0	1
	Frame Counter	776	0	2

Table 4.4 TIROS Sensitivity Results (continued)

		Kind	Sensitivity	Usage
Mission Peculiar				
I	Calib. Source	135	1	1
	Shutter	173	1	1
	Calib. Ramp. Logic	676	1	1
	In Flight Calib. Unit	716	1	1
	In Flight Calib. Logic	773	1	1
S	Primary Mirror	108	1	1
	Secondary Mirror	109	1	1
	Collimating Lens	114	3	3
	Feed Horns	121	2	2
	Polarization Splitter (OMT)	125	2	2
	Light Pipe	133	3	3
	Channel Optics	146	2	3
	Filter & Focus Lens	147	2	2
	Fixed Filter	148	17	17
	Timing Logic	150	1	1
	Gas Chamber	162	3	3
	Ant Motor Shaft	163	1	1
	Ant Pulley Drive	164	2	2
	Dicke Switch	166	4	4
	Local Oscill	167	8	8
	Proton Telescope	169	2	2
	Electron Telescope	170	2	2
	Particle Telescope	171	1	1
	Chopper Mirror Dr	612	1	1
	Motor Drive	615	3	3
	Solid State Photo Detector (Vis)	631	12	12
	Solid State Photo Detector (IR)	632	10	10
	Chopper	646	1	1
	Earth Shield	671	1	1
	Patch Heater	673	1	1
	Motor Logic	674	1	1
	Scan Logic	675	1	1
	Aux Scan Logic	677	1	1
	Primary Optics	683	7	7
	Secondary Optics	684	7	7
	Signal Conditioner	686	16	16
	CTL Logic	687	1	1
	Chopper/Driver Logic	688	1	1
	Step/Scan Dr Logic	689	1	1
	Scan Position Pickoff	692	1	1
	Optical Cell	694	3	3
	PMC Data Select	695	3	3
	PMC Freq. Control	696	3	3

Table 4.4 TIROS Sensitivity Results (continued)

		Kind	Sensitivity	Usage
Mission Peculiar				
S	PMC Drive Coil & Mag.	697	3	3
	PMC Drive Amp.	698	3	3
	Phase Shift Detector	699	3	3
	Integrator Circuit	700	3	3
	Main Programmer	702	3	3
	Ant. Reflector & Bearing	705	2	2
	Dicke Switch Driver	706	4	4
	Mixer & Preamp	708	4	4
	IF Amp. & Detec	709	4	4
	Video Amplifier	710	4	4
	Phase Detector	711	4	4
	Xtal Monitor	714	4	4
	Signal Analyzer	715	3	3
	Static Deflec Plate	717	4	4
	Channeltron	718	4	4
	Channeltron Preamp	719	4	4
	Proton Detector	721	9	9
	Electron Detector	722	2	2
	Cerenkov Radiator	725	1	1
	Signal Conditioner	686	16	16

NOTE: Kind numbers refer to component number on GO diagrams

P = Power Conditioning Hardware
D = Data Handling Hardware
C = Command Handling Hardware
I = In-flight Calibration Hardware
S = Sensor Hardware

4.2.4 HEATE-1

Conceptual block diagrams of the HEATE-1 payload are depicted on Drawings D1 & D2. These drawings show the essential components and functional flow that make up the sensor configurations proposed for HEATE-1.

HEATE-1 represents a possible concept for a Temporal X-Ray Explorer mission. This mission contains two primary sensors, i.e.,

X-RAY SURVEY CAMERAS (XSC)

TEMPORAL RESOLUTION COUNTERS (TRC)

These drawings should be compared with the equivalent "GO" logic charts found on drawings D3 & D4.

<u>XSC</u>:

There are four X-ray Survey Cameras on board HEATE-1. They are mounted to provide continuous coverage of the hemisphere as viewed on local zenith. Each camera is made up of a small window (Signal 1 in Supertype 127), which admits X-rays to a large, square, proportional counter located immediately below the window. The proportional counter contains 32 resistive anodes (Supertypes 125 & 127) each of which can be resolved by analog electronics into 32 one cm elements, yielding 1024 separate proportional counters. (Signals 2-10 in Supertypes 127). This arrangement permits direction cosine location of any source in the sky to a theoretical accuracy of 0.6 mr. The analog electronics consists of a number of charge sensitive preamps, one on each end of the 32 resistive anodes, for a total of 64 preamps. These preamp outputs (Supertype 126) are fed into an x-y data encoder, (Supertype 127). The coded coordinates are placed in a micro-processor memory for later retrieval by the S/C data processor (Signals 435-437). An anticoincidence circuit inhibits data processing in the presence of a bremsträhl ung shower or noise triggering.

TRC:

The Temporal Resolution Counters (Supertype 126) consist of six collinated proportional counters feeding a summing amplifier, a pulse shape discriminator, and a level threshold detector in parallel, (Signals 428, 420-425). An anti-coincidence circuit (Signals 419 & 426) operates to negate the output for bremstrahlung showers or for internal noise triggering. The buffered output is brought out to the same data processor as used for the X-ray Survey Cameras. Temporal time tags are tied to each event with one microsecond accuracy. A Polaris type star tracker and magnetometer is used to define the inertial position of the S/C with respect to the Celestial Equator. Commands can be used to drive the S/C to any prearranged attitude using the coordinate processor and gimbal drive systems, (Signals 440-453).

DATA HANDLING:

HEATE-1 will require unregulated S/C power, logic power supplies for the digital circuitry, regulated power for the analog circuitry, and high voltage power for the various proportional counters. GO drawing D3 depicts the major portion of these power supplies. It will be noted that all of these power supplies are doubly redundant, i.e., once after the DC/DC Converter, and again at the output for each supply. The exception to this is with the RPS power units, which are singly redundant, after the DC/DC converter unit. This was done to evaluate the effect on sensitivity.

SENSITIVITY & USAGE:

The sensitivity and usage of components list in HEATE-1 has been combined with the same information from HEATE-2. See Table 4.5 at end of Section 4.2.5.

4.2.5 HEATE-2

Conceptual block diagrams for the HEATE-2 payload are shown on drawing E1 & E2. These drawings depict the essential components and functional flow that make up the sensor configurations proposed for HEATE-2.
HEATE-2 represents a possible concept for a Transient Gamma-Ray Burst Explorer Mission. This mission contains four primary sensors, i.e.,

GAMMA SPECTRUM ANALYZER (GSA)
GAMMA RAY VECTROMETER (GRV)
CHARGED PARTICLE DETECTOR (CPD)
POLARIMETER & TIME RESOLTER (PTR)

These drawings should be compared with equivalent "GO" logic charts found on drawings E3 & E4.

GSA:

The Gamma Spectrum Analyzer is a high resolution instrument consisting of two identical detector arrays, each one incorporating three NaI detector assemblies and a Gamma source for calibration reference. The elements of each array are monitored by a set of photomultiplier tubes and post detection signal electronics, (Supertypes 128 & 129).

GRV:

The Gamma Ray Vectrometer is designed to provide precise directional information for gamma burst sources. It consists of a cubicle array of identical CsI wafer detector elements imbedded in a large CsI cubical anticoincidence shield. Each of the detector elements are monitored by a set of photomultiplier tubes and associated signal processing electronics (Supertypes 131, 132 & 133). The Vectrometer also monitors the background count in the cubical anticoincidence shield. An anticoincidence circuit is used to inhibit Vectrometer output if bremstrählung or similar noise triggering should occur.
Since the response of a particular detector elements to a

given gamma burst is proportional to its projected area normal to burst source direction, the direction cosines of the burst can be readily resolved by comparing the relative count rates in any three orthogonal detectors.

CPD:

Two solid-state detector arrays are provided to obtain independent confirmation of charged particle fluxes. It is planned that each of the arrays use solid-state charge preamps and associated signal processing electronics, (Supertype 130).

PTR:

The Polarimeter and Fast Time Resolver instrument is composed of two identical detector assemblies. Each assembly is made up of an array of seven plastic scintillators with associated photomultiplier detectors and signal processing electronics (Supertypes 134 & 135). The output of this instrument is also inhibited if random coincidence triggering conditions are encountered. The extremely fast time response of plastic scintillators makes possible the recording of gamma burst histories with microsecond time resolution and pulse-to-pulse separations down to 20 nanoseconds. By using the coincidence logic, it is possible to detect scattering asymmetries that will occur if the gamma ray photons are polarized.

All data for HEATE-2 is channeled into a central science processor which has a burst keyed memory unit to record the large amount of data which accumulated during a burst event. The burst memory is read out at a slower rate after the burst event has finished and the data is transferred to output buffers. (Signals 321-333).

DATA HANDLING:

As discussed above, data from all sensors on HEATE-2 will be processed by a dedicated science processor which contains a burst keyed memory as an integral part of the processor.

It is possible that this function can be handled by a small micro-processor or micro-computer. After initial processing, the normal data handling rate for HEATE-2 is not expected to exceed 2 Kbps including all Engineering & Housekeeping data. This is easily handled by the S/C Telemetry RIU unit. The command data rate is expected to be less than 2 Kbps.

POWER CONDITIONING:

HEATE-2 will require S/C 28V unregulated power, a logic power supply for digital circuits, a regulated voltage supply for the analog circuitry and a high voltage supply for all the photomultiplier tubes used. In GO diagram E3, all power supplies except the regulated power supply (RPS) have been assumed to be doubly redundant. The RPS supply for HEATER-2 is assumed to have a single redundancy as was done for HEATE-1.

SENSITIVITY & USAGE:

Table 4.5 itemizes the results of the functional sensitivity evaluation for HEATE-1 and HEATE-2 by various hardware categories and also indicates the 'kind' number assigned to each given kind of component on the GO logic diagrams. The table also lists a usage number which defines the number of times a given component is used in the onboard sensors for any given flight of that mission.

Table 4.5 HEATE 1 and 2 Sensitivity Results

		Kind	Sensitivity (1)	Sensitivity (2)	Usage (1)	Usage (2)
Mission Common						
P	EMI Filter	101	2	2	2	2
	Clock Regulator	105	1	1	1	1
	Power Lines	126	252	64	440	170
	Voltage Regulator	128	0	39	44	50
	Power Filter	129	0	39	44	50
	DAC	138	0	0	20	14
	DC/DC Converter	140	0	0	36	30
	Digital I/F Unit	141	0	0	20	14
	HV Multiplier	142	0	0	20	14
	HV Regulator & Filter	143	0	0	20	14
	Clock	601	0	0	2	2
	Error Amplifier	667	0	0	20	14
	Amp. Control Oscill.	668	0	0	20	14
D	Clock Regulator	105	1	1	1	1
	S/C Telemetry	131	1	-	1	-
	Output Register	185	4	-	4	-
	Clock	601	0	0	2	2
	RIU	602	1	1	2	1
	S Band Transponder	629	1	1	1	1
	Tape Recorder	630	1	1	1	1
	Analog. Multiplexer	640	1	1	1	1
	Sample & Hold Circuit	641	-	1	-	1
	ADC	642	1	2	1	2
	Command Switch	662	0	13	36	30
	Digital Multiplexer	701	1	-	1	-
	PCM Control	735	-	1	-	1
	Bit Rate Circuit	737	-	1	-	1
	4 High Level Add. Mux.	740	-	1	-	1
	2 Digit Mux	741	-	1	-	1
	Arithmetic Processor	745	4	-	4	-
C	Command Receiver	106	1	1	1	1
	Central Computer	107	1	1	1	1
	RIU	602	2	1	2	1
S	Post Amplifier	672	-	2	-	2
	Veto Matrix ckt	772	1	-	1	-
	Anti Coincidence ckt	730	5	3	5	3
	Gen Purpose Discriminator	731	-	31	-	31
	Stretch Amplifier	732	-	8	-	8
	Threshold Discriminator	733	1	8	1	8
	Timing & Scaling amp	734	-	31	-	31
	Torque Motor	756	2	-	2	-
	Pulse Shape Discriminator	750	1	-	1	-
	Summing amp	749	1	-	1	-

Table 4.5 HEATE 1 and 2 Sensitivity Results (continued)

		Kind	Sensitivity		Usage	
Mission Peculiar			(1)	(2)	(1)	(2)
D	Frame Counter	736	-	1	-	1
I	Calib. Source	135	-	2	-	2
S	Antenna Gimbal	122	2	-	2	-
	N_aI Detector	180	-	6	-	6
	C_sI Detector	181	-	5	-	5
	Plastic Scintillator	182	-	14	-	14
	Spark Grid Window	183	4	-	4	-
	Proportional Counter Collimator	184	6	-	6	-
	Programmable Memory	186	4	-	4	-
	C_sI Cubic Shield	187	-	1	-	1
	Detector Preamps	634	252	2	256	2
	Main Programmer	702	-	1	-	1
	HEATE-2 Memory	738	-	1	-	1
	Science Processor	742	-	1	-	1
	x-y spark processor	743	4	-	4	-
	x-y decoder	744	4	-	4	-
	HEATE-1 Memory	748	4	-	4	-
	Readout Sensor	755	2	-	2	-
	Charged Particle Detec	757	-	2	-	2

NOTE: Kind numbers refer to component number on GO diagrams

P = Power Conditioning Hardware
D = Data Handling Hardware
C = Command Handling Hardware
I = In-flight Calibration Hardware
S = Sensor Hardware

4.2.6 GRE

Conceptual block diagrams of the GRE payload are depicted on drawings F1 & F2. These drawings show the essential components and functional flow that make up the sensor configurations proposed for GRE. GRE represents an alternate concept for a Gamma Ray Explorer Mission. This mission contains four basic subsystems each utilizing different sensors, i.e.:

SPARK CHAMBER SYSTEM (SCS)
TOTAL ABSORPTION SHOWER COUNTER (TASC)
ANTICOINCIDENCE DOME SYSTEM (ADS)
SCINTILLATOR TIME-OF-FLIGHT SYSTEM (STFS)

These drawings should be compared with the equivalent GO logic charts found on drawings F3 & F4.

SCS:

The Spark Chamber System consists of a stack of 36 spark modules, interleaved with an equal number of pair production/scatter plates. The SCS contains high voltage pulsers, two arrays of plastic scintillator light-pipe photomultiplier tube assemblies and two hermetically sealed units containing programmable power supplies and triggering circuitry. The Spark Chamber is divided into an upper & lower section. In the upper section, there are 24 close spaced Spark Chamber modules interleaved with tantulum pair-production plates. Each module has a 32x32 wire grid. Each wire in the grid is threaded through a magnetic core. The lower section is similar. The remaining 12 spark modules are spaced equally between the upper and lower plastic scintillation tiles and are interleaved with tantulum scattering plates, (see Supertype 126 & Spark Chamber counter drawing F3).

In practice, a high energy gamma ray incident within the aperture of this instrument will convert to a positron-negatron pair in one of the tantulum pair-production plates.

The newly formed electron pair then propagate downward through the Spark Chamber, triggering at least one counter in each of the two 3x3 plastic scintillation tiles. (see Supertype 127 & T-O-F Analyzer drawing F3). In the absence of a signal from the large plastic scintillator, anticoincidence dome, enclosing the top of the sensor, the signals from the two T-O-F scintillator arrays cause the generation of a master-event signal, (see Supertype 128 & anticoincidence dome drawing F3). The master-event trigger (MET) signal initiates Spark Chamber firing, thereby recording the trajectories of the electron pair particles and, through momentum calculations, the arrival direction of the original gamma ray can be ascertained.

The MET signal also initiates an anlysis of the incoming signal with the Total Absorption Shower Counter (TASC), which consists of a large NaI (Tℓ) crystal scintillator using twelve photomultiplier tubes. Since the energy of the electron pair is dissipated in this detector, the TASC event provides a precise determination of the energy of the original gamma ray, (see Sypertype 127, & TASC on drawing F3).

The pair trajectory path, as defined by ionized gas molecules, is recorded by the magnetic cores on each spark module when the MET signal is received. Shortly after the setting of these cores, the cores are sequentially examined in a destructive read-only mode. Drive pulses are used to interrogate the state of each magnetic core, and the resulting information is passed on by an array of sense amplifiers tied to each line of the wire grids.

DATA HANDLING:

The data from all events is fed into a dedicated processor similar to the one used on HEATE-1, and into a buffer memory. The data is read out of memory at a slower rate into the S/C telemetry system. The readout rate for GRE is not expected to

exceed Kbps which can be handled readily by the S/C telemetry RIU. Engineering & Housekeeping data are included in the above rate.
Command rates are not to expected to exceed 2 Kbps.

POWER CONDITIONING:

Power for GRE is provided by S/C 28V unregulated, logic power for digital circuits, regulated power for analog circuits, and high voltage power for the photomultiplier tubes. All power supplies are doubly redundant except for the RPS, which is singly redundant as described for HEATE-1 & HEATE-2, (see Signals 101-104 on drawing F3).

SENSITIVITY & USAGE:

Table 4.6 itemizes the results of the functional sensitivity evaluations on GRE by various hardware categories and also indicates the 'kind' number assigned to each component on the GO diagrams. A usage number is also given, which defines the number of times a given component is used in the onboard sensors for any given flight of that mission.

4.2.7 SEASAT

A conceptual block diagram of the SEASAT payload is depicted on drawing G1. This drawing shows the essential components and functional flow that make up the sensor configurations postulated for the advanced SEASAT mission.

This mission contains six basic sensors, i.e.:

ALTIMETER, RATIO (ALT)
SCATTEROMETER, MICROWAVE (SCAT)
SYNTHETIC APERTURE RADAR (SAR)
VERTICAL IR RADIOMETER (VIRR)
SCANNING MULTICHANNEL MICROWAVE RADIOMETER (SMMR)
TRANET BEACON (TRAN)

Equivalent GO logic diagrams for SEASAT were not made up during this study because of the highly integrated design

Table 4.6 GRE Sensitivity Results

		Kind	Sensitivity	Usage
Mission Common				
P	EMI Filter	101	2	2
	Clock Regulator	105	1	1
	Power Line Leads	126	45	172
	Volt Regulator	128	27	44
	Power Filter	129	27	44
	DAC	138	0	18
	DC/DC Converter	140	0	32
	Digital Interface Circuit	141	0	18
	High Voltage Multiplier	142	0	18
	HV Regulator & Filter	143	0	18
	Clock	601	0	2
	Error Amplifier	667	0	18
	Amp. Control Oscillator	668	0	18
D	Clock Regulator	105	1	1
	Clock	601	0	2
	RIU	602	1	2
	S Band Transmitter	629	1	1
	Tape Recorder	630	1	1
	Analog Multiplexer	640	2	2
	Sample & Hold	641	1	1
	ADC	642	1	1
	Command Switch	662	10	33
	Digital Mux	701	2	2
	S/C Telemetry	728	0	1
C	RIU	602	1	2
	Command Reciever	775	1	1
I	LED	191	30	54
S	Magnetic Core	189	40	40
	Counter Gas Cylinder	190	2	2
	Gen. Purpose Discriminator	731	0	1
	Summing Amplifier	749	1	5
	Coincidence Logic	774	1	1
	Event Flag Gate	183	0	1
	Position Encoders	744	40	40
	Sense amps	759	40	40
	Co[illegible] Driver	761	40	40
	Trigger Out	762	1	1
	R/O Electronics	763	1	1
	Pulse Height Discrim.	765	1	1
	Time Logic	766	1	1
	Time Encoder	767	1	1

Table 4.6 GRE Sensitivity Results (continued)

		Kind	Sensitivity	Usage
Mission Common				
S	Time of Flight Logic	768	1	1
	Coincidence Matrix	769	1	1
	Direc. Cosine Logic	770	1	1
	Met Trigger ckt	771	1	1
	Veto Logic ckt	772	1	1
Mission Peculiar				
D	Memory	738	0	1
	Frame Counter	776	0	2
I	In Flight Calib. Logic	773	1	1
S	N_aI Detector	180	1	1
	Plastic Scintillator Tiles	182	2	2
	Pair Production Plates	188	40	40
	Dome Scintillator	192	1	1
	Wire Grid Matrix	758	40	40
	HV Pulser	764	0	2

NOTE: Kind numbers refer to component number on GO diagrams

P = Power Conditioning Hardware
D = Data Handling Hardware
C = Command Handling Hardware
I = In-flight Calibration Hardware
S = Sensor Hardware

already evidenced in the SEASAT system, i.e., the Data Handling and Power conditioning support systems in SEASAT appear to follow the overall pattern being recommended for the previous systems considered by this study.

<u>ALT</u>:

The Radio Altimeter to fly on SEASAT is a high precision model capable of measuring relative wave heights and satellite altitudes to 10% or 0.5 m. The science data will be transmitted at approximately an 8.2 Kbps rate.

<u>SCAT</u>:

The SEASAT Scatterometer is an active instrument used to monitor sea state and wind speed when rough water is present. The science data will be transmitted at a 590 bps rate. However, data read-in rates may be as high as 85 Kbps.

<u>SAR</u>:

The Synthetic Aperture Radar instrument will be used to obtain differential correlation data between various sea/land features, e.g., deep ocean and near coastline wave spectra, bare land and snow covered land, sea ice & fresh water ice, etc. Science data will be transmitted to ground over a dedicated SAR Data Link. Engineering data will not exceed 500 bps.

<u>VIRR</u>:

The Vertical IR Radiometer is being used to obtain visual & IR imagery of the ocean surface and associated cloud coverage, to derive ocean surface temperatures. The science data will be multiplexed at a rate of 12 Kbps.

<u>SMMR</u>:

The Scanning Multichannel Microwave Radiometer is a passive system which will confirm ocean surface, wind speeds, rainfall, etc., to a high degree of accuracy. Science data from the SMMR will be processed at a 2 Kbps rate.

DATA HANDLING:

Present designs for SEASAT assume that science & engineering data from all sensors except SAR will be passed to a central processing unit and transmitted via a common S-band transponder. Overall transmission data rates are quite compatible with the S/C telemetry RIU planned for use on the MSS. The command data rate required to track & control all operating sensors on SEASAT is not expected to exceed 2 Kbps.

POWER CONDITIONING:

Present SEASAT design calls for dedicated power supplies for each sensor, each supplied by the S/C 28V unregulated power. A review of the other missions described in this section indicate that High Voltage Power Supplies, and probably most of the Regulated Power Supplies for analog circuitry are best dedicated to each sensor in accordance with power required, while the logic power supply and unregulated power supply are best centrally located.

SENSITIVITY & USAGE:

There are no sensitivities or usage numbers for the SEASAT mission.

4.3 Potential Standard Functions and Standard Interface Modules

4.3.1 Potential Standard Functions

4.3.1.1 Power Conditioning

A cursory inspection of the sensitivity tables in the preceeding section indicate that the common hardware function reflecting the greatest overall usage is power conditioning hardware. Since many of the components in this category were assumed to be nonredundant, the high usage per mission flight also results in relatively high sensitivity numbers. The effects of using redundant power units can be seen most easily by looking at the sensitivity charts for LANDSAT and TIROS, e.g.: In the Power Conditioning category for LANDSAT, Table 4.2 lists a sensitivity of 3.0 for the high voltage passive network feedback. The high voltage supplies in LANDSAT are not redundant. In the TIROS system (Table 4.4), the high voltage supplies were assumed to be redundant, and the sensitivity values listed for the high voltage multipliers, and components of the active feedback net, such as error amp's, amplitude control oscillators, etc. are essentially zero.

Consider the power supply components for GRE, Table 4.6. All power modules except the Regulated Power Supply (RPS) were assumed to be doubly redundant. The RPS supplies were assumed to be singly redundant after the DC/DC converters, hence the DC/DC converters show a sensitivity of zero. However the RPS power supplies possess no redundancy for components such as voltage regulators, power filters, and the many terminal leads or attachment points after the DC/DC converters. This is reflected in the relatively high sensitivity numbers for these components. It is evident that overall system reliability will be significantly increased if the various power supplies that are required are designed with inherent redundancy at the output.

4.3.1.2 Data Handling

The next function reflecting relatively high usage from mission to mission are components in the data handling category, particularly data multiplexers (analog and digital), sample and hold circuits, analog-to-digital (ADC) or digital-to-analog (DAC) conversion units, and output buffers or registers. Since data handling requirements vary quite widely from one sensor to another, this area needs to be examined closely. An ideal situation would be one in which only software changes were needed to handle the wide variety of data rates and multiplexing or formatting requirements. This would minimize the number of hardware configurations required. It would also be ideal if the components or functions required for a data handling system could be standardized so that any given data handling system could be expanded or contracted by simply adding or removing components like building blocks. Whenever possible, each NASA flight should have a standby data handling unit (DHU), which can be commanded on if the primary DHU should fail. NASA has recently defined a standard Remote Interface Unit (RIU) to process telemetry data. The Telemetry Interface RIU is designed to fly on the Multi-Mission Modular Spacecraft (MMS). This RIU will contain a standard multiplexer with 64 inputs which can be used for analog, bilevel, or serial digital signals. The number of multiplexer inputs can be expanded in groups of 64 by adding additional RIU's. All analog inputs are digitized to 8 bit words. The output capacity of the RIU is 64 kbps. It was assumed for this study that all missions having science and engineering data rates less than 64 kbps would make use of the MMS Remote Interface Unit Telemetry Interface. Rates higher than 64 kbps will require an update of the current RIU capability.

In addition, many of the planned sensor configurations have special signal processing requirements e.g.:

a. Data output may be at a higher rate than normal state-of-the-art can handle, hence special gating and data division schemes must be employed.

b. Special signal characteristics may be needed, such as the time derivative of a given signal or the integrated value of a selected channel over a given time period, etc.

c. Data with extremely high input rate may have to be recorded or stored in a temporary memory buffer and read out at slower rates for the telemetry.

These special signal processing requirements are usually met by using hardwired memory/buffers, or special micro processors with random access memories. An ideal data handling unit would be expected to incorporate or provide capability for as many of these features as is feasible.

4.3.1.3 Inflight Calibration

A common function from mission-to-mission is the necessity for inflight calibration. This requirement has been treated as a sub-function of the data handling function. This function is usually nonredundant, hence the nonzero sensitivities, and the kind of components required usually vary from one kind of detector system to another. It is not recommended that this function be standardized due to its relatively low usage. However, it is recommended that NASA consider setting up a calibration standards panel to define acceptable standard calibration sources for each type of detector in common usage. In addition, it is recommended that serious attention be given to each sensor design with the view of determining the feasibility of making key parts of the calibration function redundant, i.e., standby calibration motors, redundant motor or lamp driver circuits, etc.

4.3.1.4 Command and Control Functions

The command and control subsystem of the spacecraft is designed to receive ground or onboard stored commands, decode them, and transmit them to the various spacecraft or sensor subsystems to perform desired functions, such as:

a. Adjustment of high voltage supplies.
b. Changing the operational mode or state of various sensors, or power supplies.
c. Resetting spacecraft or sensor clocks.
d. Changing telemetry modes and formats, etc.

NASA has defined a standard Remote Interface Unit (RIU) to process command data. The Command Interface RIU is designed to fly on the

Multi-Mission Modular Spacecraft (MMS). This RIU contains a command decoder with 64 discrete command outputs and eight serial magnitude command outputs. Each sensor requiring commands will be tied directly to an assigned RIU channel. Command rates up to 64 kbps can be handled by this unit. For the present time, these RIU's make additional standardization in the command and control area unnecessary.

4.3.1.5 Sensor Hardware

The last common hardware group to be examined for potential standardization opportunities are the sensor hardware components. Table 4.7 lists components falling into this category that show relatively high usage either within a given mission or across several missions.

It will be noted that six of the high usage components as listed in Table 4.7 are each used on one mission only. At present, these are one flight missions. Unless there are several more missions or flights using similar components, the usage is still a one time only usage and does not justify standardization.

Two of the components listed in Table 4.7 form elements of a supporting subsystem, i.e., the temperature monitoring circuits and system heaters. These components are used on all earth observation missions each of which have several flights each. However, the heater sizing and operating regions for temperature monitors are usually set by the type of orbit (polar, equatorial, sun synchronous, geosynchronous, etc) and the thermodynamic characteristics of each payload. These vary enough so that standardization benefits are meager. However some attention should be paid to the possibility of making these items redundant to increase reliability.

Table 4.7 Sensor Components with High Usage

Component	Usage	No. of Missions	Mission
Magnetic core, Sense Amp & Driver	40	1	GRE
Position encoders	40	1	GRE
Timing & Scaling circuits	31	1	HEATE 2
General purpose discriminators	32	1	HEATE 2
Temperature monitoring circuits	26	3	STORMSAT, TIROS, LANDSAT
Detector post amplifiers	24	2	TIROS, HEATE 2
System heaters	18	3	LANDSAT, TIROS, STORMSAT
Step/scan motors & drivers	10	3	STORMSAT, TIROS, LANDSAT
Coincidence/Anti circuits	9	3	HEATE 2, HEATE 1, GRE
Threshold discriminators	9	1	HEATE 2
Scan position resolvers	8	3	STORMSAT, TIROS LANDSAT
Stretch amplifiers	8	1	HEATE 2
Summing amplifiers	6	2	HEATE 1, GRE
Focus drive	4	2	STORMSAT, LANDSAT
Optical filter motors & drives	3	2	STORMSAT, LANDSAT

The remaining components in Table 4.7 form integral parts of the sensor circuitry to which they belong, i.e., scan motor and drives, focus drives, optical filters and drives, position resolvers, and detector post amplifiers. They are used on a number of missions with relatively high frequency. However mission operation requirements, scan frequency, operational altitude, depth of field, etc., vary sufficiently from mission to mission so that the benefits of standardizing these items seem to be minimal. When specifying interfaces for standardization, it is desirable to maintain clearcut interfaces to extend the MMS interface in a logical manner up to the experimenter's equipment. It does not seem desirable to piece-meal the sensor subsystems with standard components for the sake of standardization. This argument applies to the items previously described in this paragraph as well as to items such as summing amplifiers and anti or co-coincidence circuits. It is recommended, however, that each of those components that contribute significantly to system sensitivity be examined with the idea of lowering that sensitivity through redundancy or other design tradeoffs.

4.3.2 Standard Interface Modules

4.3.2.1 Power Conditioning Modules

Traditionally, spacecraft and sensor designers have built their own power supplies so that they could build in the desired regulation and protective features they felt were necessary to optimize equipment performance. If standard power modules were available, these modules could be provided as GFE to approved experimenters at the beginning of the sensor hardware design stage. Under these conditions, the experimenters need only add whatever additional filtering or protective features that they feel is necessary to be compatible with the standard power modules. The added components would then be treated as mission specific or mission peculiar items. The standard power units which are to be recommended are based on the results of this selective study. This study includes some 27 independent sensors and 7 distinct missions which were evaluated in depth and another 44 sensors on 7 missions which are related to the first set in terms of kind of sensors and approximate power needed, and kind of data processing required.

A review of the power needed by the sensor systems indicates that a minimum of four basic power module types will satisfy almost all requirements. Each of the four standard types are proposed as dual cross-strapped units to increase system reliability. In practice, each single unit of a dual unit system will furnish sufficient power to take care of all equipment normally attached to the dual unit. In normal operation it is expected that one half of each dual unit will be placed on command standby, and will be turned on only if the operating unit should fail or if overall power consumption becomes too high for a single unit to supply. The reference design for each of the standard power modules is based on hardware which is presently available. Table 4.8 lists typical available weight and volume characteristics for components which might be used in the design of standard power modules. Additional weight and volume has been added to these numbers to allow for overall mechanical closure and packaging in a manner which would be analogous to NIM type specifications. NIM specifications allow a wide degree of mechanical and electrical interchangeability for all types of transistorized modular instruments, power supplies, connectors and mounting bins. The recommended standard power modules are as follows:

1. STANDARD HIGH VOLTAGE SUPPLY (HVS)

   High voltage, low current, programmable supplies are needed for those sensors using photomultiplier tubes, static deflection plates, channeltron analyzers, proportional counters, vidicons, etc. The voltage requirements differ widely for these different sensors. However, in the missions selected for this study, the need for photomultiplier voltage supplies was by far the most frequent. The other sensor requirements for high voltage are believed to be infrequent enough to not warrant standardization at this time. Recognizing this, it is recommended that the photomultiplier high voltage supply be designed so that it can be modified or adapted with a minimum of change to be compatible with the other high voltage applications. A typical standard module satisfying these requirements is shown in Figure 4.1.

   To minimize power arcing and breakdown at these high operating voltages in space environments, the high voltage supplies

Table 4.8 SIM Component Data (Power Conditioning)

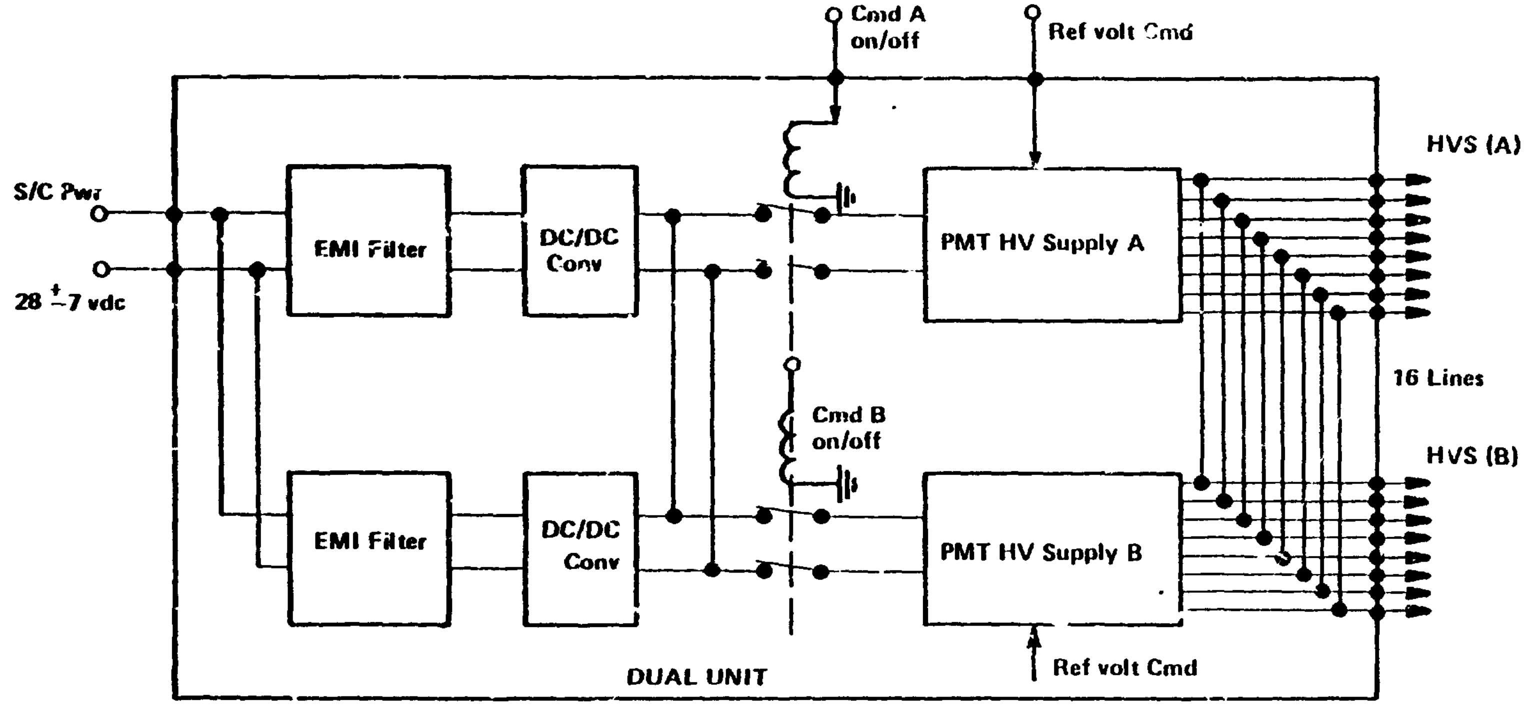
Component	Weight (grams)	Volume (cm^3)
EMI Filters	Negligible	Negligible
Filter Pins	Negligible	Negligible
Ferrite Beads	Negligible	Negligible
Shunt Capacitors	Negligible	Negligible
DC/DC Converters	85 to 170	6.6-32.8
Command Relays		
1 Watt	28 to 56	2.6
3 Watt	28 to 56	2.6
18 Watt	170 to 227	35.6
Op-amp Oscillator and Relay	340	41.0
Connectors, Electrical	340	13.1
PMT High Voltage Multiplier	340	49.2

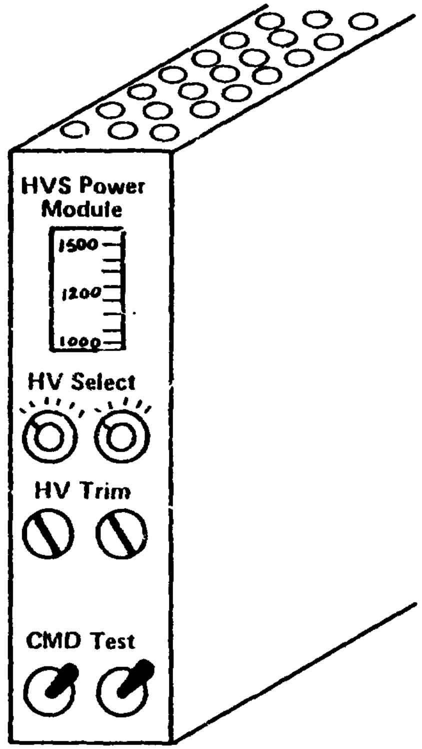
are normally pressurized and mounted as close as possible to the using elements. This requires that these modules be located next to each sensor requiring their use.

2. STANDARD LOGIC POWER SUPPLY (LPS)

Logic power supplies (LPS) are required for the digital processing part of each sensor. To conserve power and minimize thermal problems, designers frequently choose logic components from any one of three families of common logic devices. In normal applications, the components from these three families have different voltage requirements. The three logic device families are as follows:

(1) Complementary metal-oxide semiconductors (CMOS) which typically have a mid-range voltage requirement of $\pm$ 10 vdc.





CHARACTERISTICS (TYPICAL)

Command programmable (± 1500v to ± 3000 vdc)
Regulated
NIM type construction
Max Power - 2 watts (ref only)
Efficiency - 70%
16 voltage outputs (dynodes and anode)

USES

PMT supplies 1-3kv
Static deflec. plates 2-4kv
Channeltron analyzers 2-4kv
Prop. counters 3-6kv
Vidicons 6-12kv

Figure 4.1 High Voltage Supply (HVS) Power Components

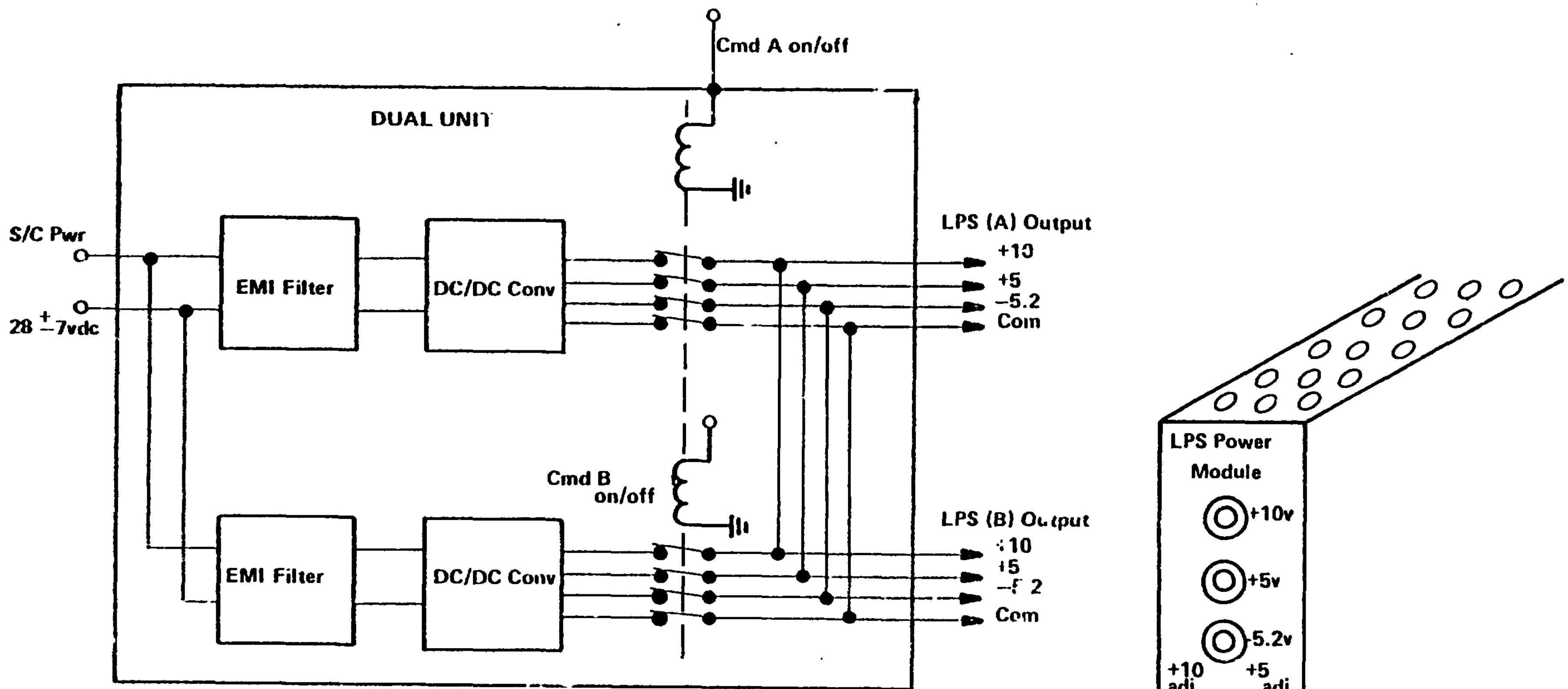
(2) Transistor-transistor logic (TTL) which typically have a mid-range voltage requirement of +5 vdc.

(3) Emitter-coupled logic (ECL) which normally has a mid-range voltage requirement of -5.2 vdc. The recommended power modules for this category should be capable of providing all three operating voltages.

Since the majority of the digital processing equipment for any given payload is usually centrally located, it seems practical to mount these modules close to the common processing equipment area and centrally distribute needed logic power to the individual experiments requiring it. Figure 4.2 depicts a typical standard LPS module.

3. STANDARD ANALOG POWER SUPPLY (RPS)

Regulated analog voltage power supplies (RPS) are required to power the large number of integrated circuit (IC) analog devices which appear to be the logical design choice for the bulk of the sensor electronics. Special devices such as switches, relays, transformers, etc, which are not of IC design, appear to have voltage specifications which are compatible with the IC components. Most IC components and special devices described will operate best in the median operating range of ± 15 volts dc. Some of the experiments evaluated used ± 12 volts and ± 6 volts in the analog devices. Most IC circuitry will operate efficiently over a fairly wide range of input values. The ± 15 volts should be capable of being used in place of ± 12 volts with no change in performance. The ± 6 volts usually represents a special bias requirement. The requirement for bias voltages can best be satisfied by providing a ± 6 volts on the power pack output and performing any additional conditioning within the sensor instrumentation that requires it.



CHARACTERISTICS (TYPICAL)
Three voltage outputs (+10v (CMOS), +5v (TTL), −5.2v (ECL))
NIM type construction
Power -25 watts (ref only)
Efficiency -70%
Regulated

Figure 4.2 Logic Power Supply (LPS) Power Components

Since almost all sensor instrumentation will require several of these power modules, it is recommended that a selected number of RPS modules be dedicated to each sensor and located physically close to the using sensors. A central distribution point for all analog power would result in considerable complication of vehicle harness wiring and some heat dissipation problems. The probability of wiring shorts or opens is also multiplied with more complex wiring, plus the increased risk of losing all payload power due to a common power system fault. A typical RPS module is shown in Figure 4.3.

4. STANDARD PULSED AND UNREGULATED POWER SUPPLY (PPS/UPS). The majority of the earth science payloads require pulsed dc power for stepping-scan motors or antennas, and unregulated dc power for heaters, special power supplies, motors, etc. In every case, the sensors or unregulated supplies use spacecraft 28 vdc as source power. The pulsed power is generally required to be synchronized by a timing pulse which operates in synchronism with the scan frequency. Since these type of modules will normally run at higher wattage values, they will generally have to have special provisions made for heat dissipation. Due to these reasons, it is recommended that these modules be located in a central area of the spacecraft, and central distribution be employed. A typical PPS/UPS module is depicted in Figure 4.4. Provisions have been made in this module to independently control unregulated power and/or pulsed power outputs through the use of independent command relays.

4.3.2.2 <u>Data Handling Modules</u>

At the present time, each sensor or group of sensors flown by NASA requires a data handling system which is tailored to the individual sensor and mission operating requirements. No doubt due consideration has been given to adapting older data handling systems to each new system, but the lack of flexibility and the large number of considerations involved has usually resulted in

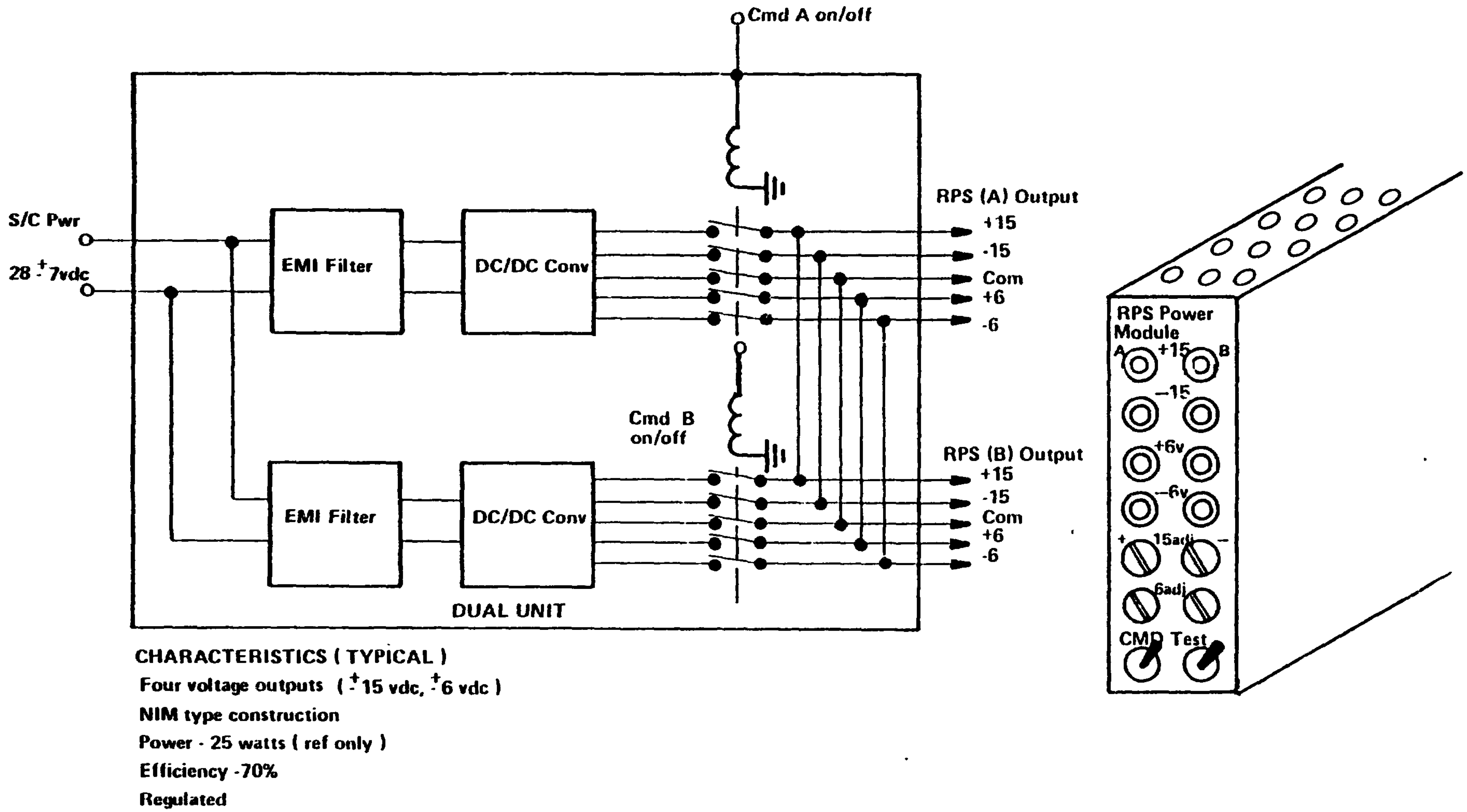
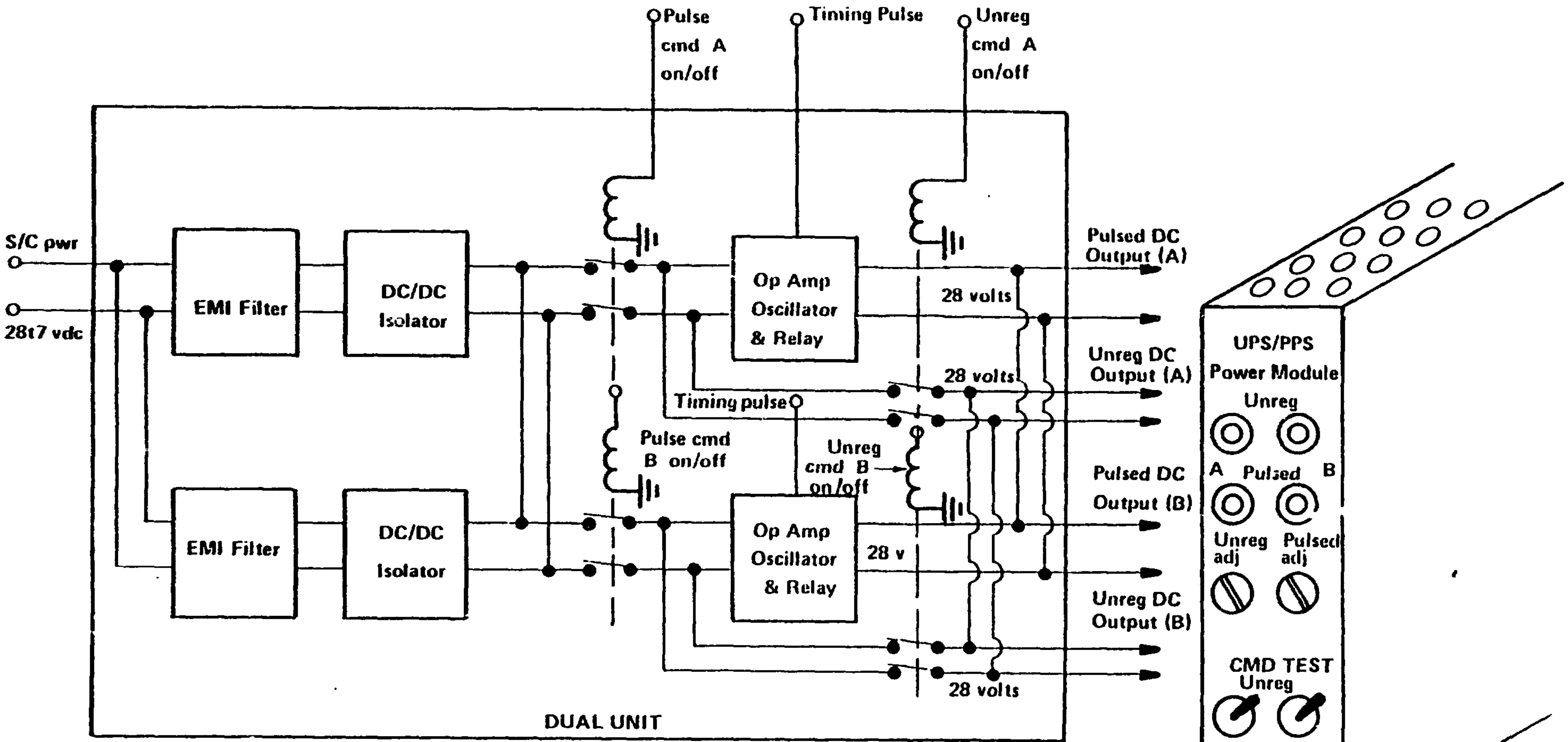


Figure 4.3 Regulated Analog Supply (RPS) Power Components



CHARACTERISTICS (TYPICAL)

Pulsed voltage - 28 vdc (step-scan motors)
Pulse rate - approximately 10-60 Hz
Unregulated voltage - 28 vdc (heaters and chopper supply
NIM type construction
Power - 100 watts (ref. only)
Efficiency - 70%
Unregulated

Figure 4.4 Pulsed and Unregulated DC Supply (PPS/UPS) Power Components the design of new data handling systems. A typical data handling system usually consists of one or more of the following components:

Analog Multiplexer
Sample and Hold Circuit
Analog to Digital Converter (ADC)
Digital to Analog Converter (DAC)
Digital Multiplexer
Buffer Memory or Output Register
Data Sequencer and Controller (programmer)

These components are often called upon to process large numbers of independent data inputs and a wide variety of data rates. Figures 4.5 and 4.6 illustrate the wide breadth of data rates that were encountered for some of the basic sensors evaluated during this study along with assumed data rates for eight related missions. The line at 64 kbps on Figure 4.5 represents the data handling capability of the present Telemetry Interfacing RIU proposed for use on the MMS. The proposed RIU does not have capability for data averaging, memory storage, delayed data readout, or etc. It cannot adequately handle the data rates shown on Figure 4.6 which are in the Mbps range. Processing of the MSS data in LANDSAT has been proposed and demonstrated by dividing the analog channels up into several parallel groups, and by using multiple Sample and Hold circuits all working in tandem. Similar approaches can be used for STORMSAT and for the Thematic Mapper on LANDSAT. However, in both STORMSAT and LANDSAT, the required basic data processing rate, after data division and multiple gating, is still well above the 64 kbps capability of the telemetry RIU's planned for use on the MMS.

In order to answer this problem and provide a capability for sophisticated data processing, short term memory storage, delayed data readout, etc., it is recommended that NASA consider an Advanced Distributed Data Processing System (ADPS) with a Standard Data/Control Bus. This approach has been successfully implemented by the Air Force for jet aircraft data monitoring and has been suggested recently by JPL to manage data on the planetary payload

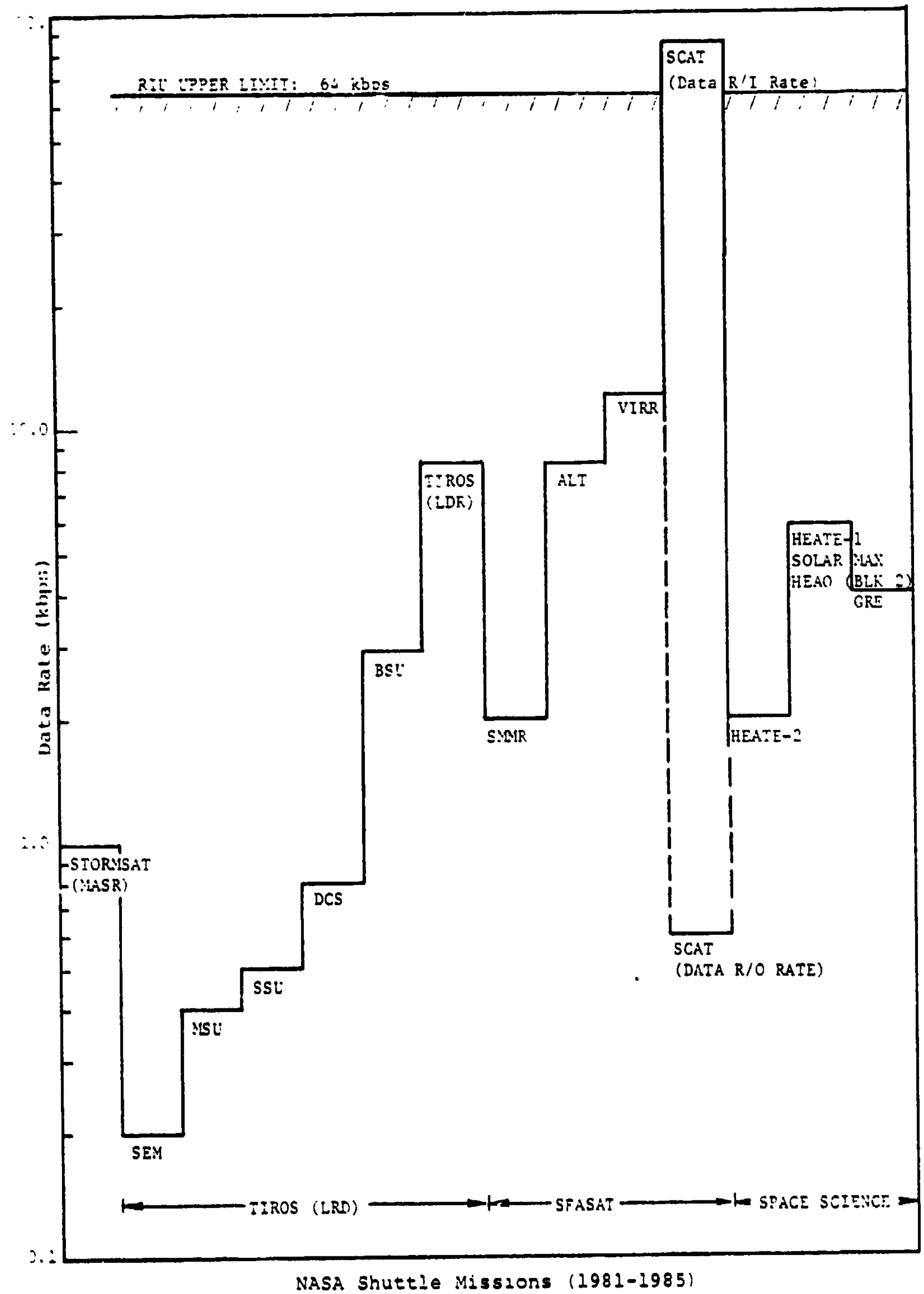


Figure 4.5 Mission Data Rate Requirements (Low)

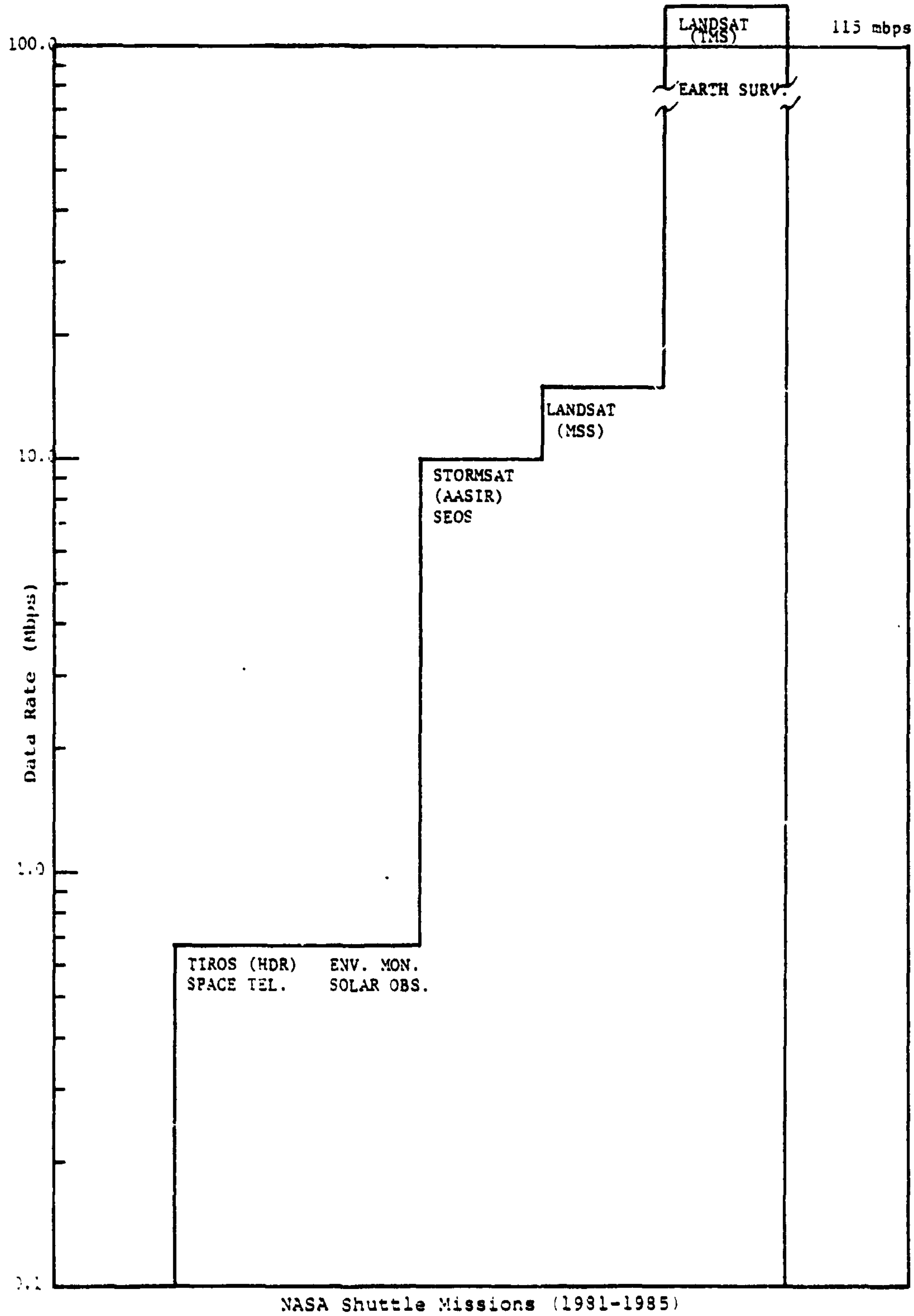
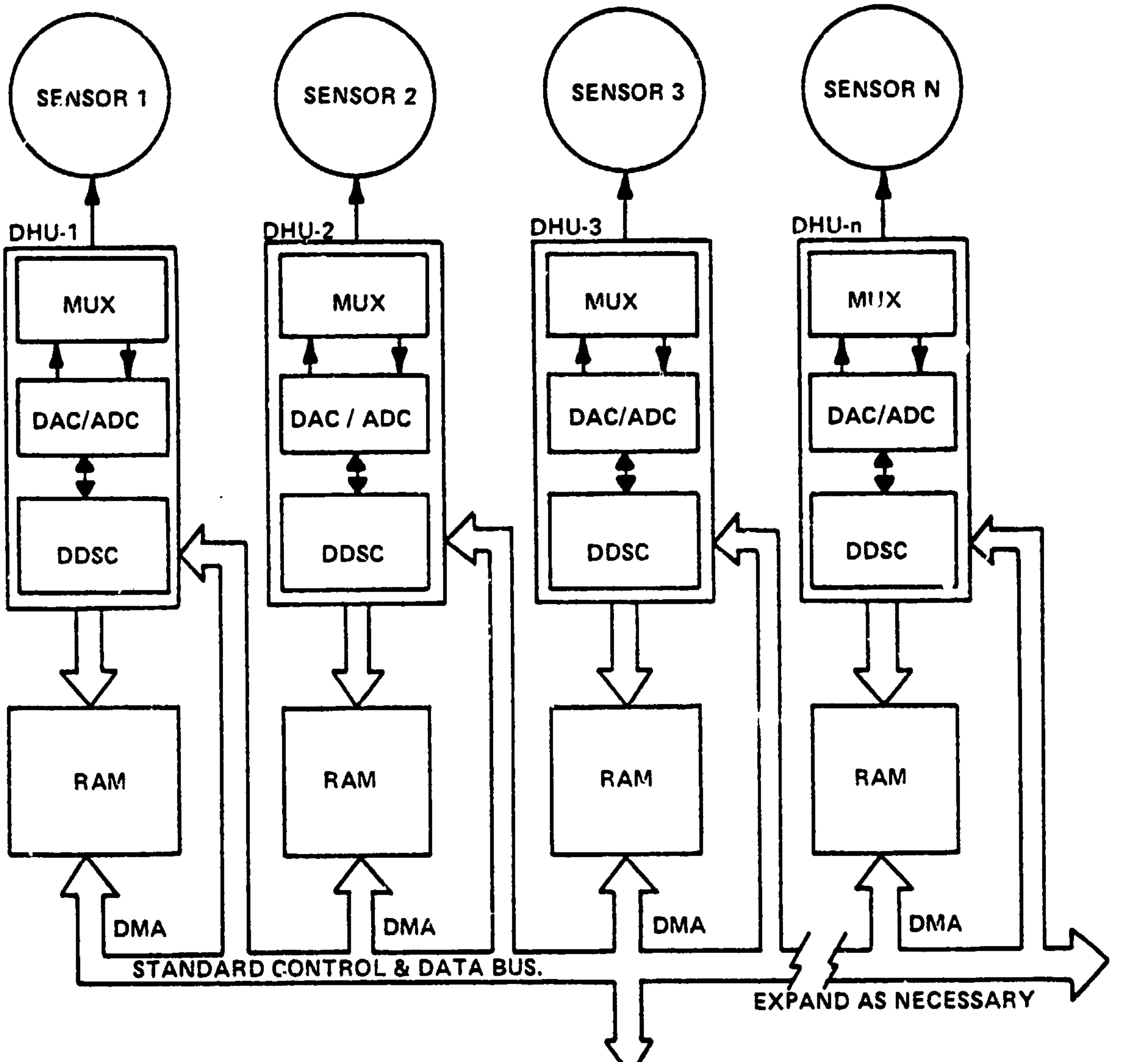


Figure 4.6 Mission Data Rate Requirements (High)

probes. A typical ADPS data system is shown schematically on Figure 4.7. In this concept, each of several Dedicated Data Sequencer/Controllers (DDSC) perform the functions of data control, data monitoring, special data processing, and data acquisition for their respective sensors. Each DDSC operates with its own Random Access Memory (RAM) for short term data storage, and its own analog multiplexer and associated digital conversion circuitry. Each DDSC with its immediate group of peripherals is designated on Figure 4.7 as a Data Handling Unit (DHU). The DDSC serves as the heart of each proposed DHU and would be constructed so that a wide range of data processing rates could be selected, either by software changes while in flight, or by hardwire changes prior to flight. The DDSC would operate in synchronism with the spacecraft clock. Items such as data formatting, sync codes, frame ID's, etc., could be predetermined from a wide choice of software or hardware options. Each of the DDSC's are in turn linked to a Master Data Sequencer and Controller (MDSC) which consists of a high speed device exercising direct memory access (DMA) control over each of the local RAM's. Direct memory access would allow a high data transfer rate from the dedicated memories to the MDSC and vice versa.

The modular design and use of a standardized bus system would provide a great deal of flexibility in the design of the various sensors and satellite payloads, while providing a common interface for data transfer, data protocol, and other direct memory access functions, such as priority interrupts. As sensors are added or deleted from a given S/C payload, the DDSC's could simply be added or deleted from the standard bus. The high impedance of the tri-state bus allows a large number of interfacing units to be added or deleted without affecting bus performance. The Dedicated Data Sequencer/Controllers could be hardwired logic, or microprocessor based systems, as necessary, to meet the operational requirements of the associated experiments.* If microprocessors are used for

*Some of the proposed sensors already include a custom microprocessor to pre-process the science data prior to formatting and storage.



TYPICAL DPU CHARACTERISTICS

48 Programmable I/O lines
Multisource priority interrupt capability
64 Byte RAM memory (12 bits/byte)
64 Byte ROM, 2μsecond response time.
CAMAC compatability

LEGEND

DDSC -Dedicated data sequencer and controller, DMA -Direct memory access
MDSC -Master data sequencer and controller
MUX -Analog and digital multiplexers
DAC/ADC -Digital to analog converter/Analog to digital converter
RAM -Random access memory (expandable as required)
ROM –Read only memory (contained in DDSC units)

Figure 4.7 Advanced Data Processing System (Distributed data processing with standard bus system) the DDSC's, each experimenter could load his own special data processing and formatting requirements into the DDSC's read only memory (ROM). If a hard wired system is used, it is recommended that special data preprocessing functions be kept to a minimum. The manner in which the DDSC's are interfaced to the standard bus system allows a large range of system operational modes. One mode might be a priority interrupt generated by one of the principal sensors upon the arrival of some important external event and passed on by its own DDSC to the MDSC. This signal would request an interrupt of all other channels so that the priority data could be processed and stored or transmitted. Another possible mode might be a command signal to shut down designated DDSC's due to a low power situation or mission task completion. The standard bus concept also offers an economical method for changing payloads from one vehicle to another in emergency situations, or for adapting previously designed payloads for new vehicles.

The Master Data Sequencer/Controller initially should be a hardwired logic system to meet the high speed data processing rate requirements and would eventually be replaced with an advanced microprocessor based system at considerable savings in system weight, cost, complexity, and increased reliability, when the data transfer rate requirements are met by rapidly advancing technology.

Reference designs for the DHU's and associated MDSC's have been based on currently available components. Table 4.9 conservatively itemizes some of the projected weight and volumes for these components. Component weight and volume values have been increased slightly to allow assembly and packaging in a manner which would be analogous to CAMAC system specifications for data handling systems. CAMAC specifications set up a general standard for data handling, operations with digital controllers or computers, and general packaging, connectors, etc. It has been assumed for this study that single channel data handling rates up to one Mbps can be handled by present state-of-the-art hardware. The DHU's and MDSC's that fall into this category are designated as "simple" units. For single channel data rates which are slightly higher than one Mbps, "simple" units can be used in parallel by adding parallel Sample and Hold circuits gated by a Master Controller. To significantly reduce the need for channel separation and parallel Sample and Hold circuits, faster DHU's and MDSC's must be used. Since the state-of-the-art in this area is advancing rapidly, it is felt that faster DHU's and MDSC's will be available in the near future. Units falling into this category have been designated as "complex" units.

4.4 Standard Interface Module Programmatic Requirements

4.4.1 Standard Power Modules

The number of standard power modules required for each sensor was checked by two independent methods, i.e.,

a. The first approach was to total up the individual sensor power requirements given in the sensor descriptions for a typical mission flight and apportion the proper number of power modules.

Table 4.9 SIM Component Data (Data Handling Unit)

Component	*Weight (kg) Per DHU	*Volume (cm^3) Per DHU	PWR (mw)
DHU	.80	983	300
MDSC	.60	655	200

*Weight and volume estimates are based on 1975 technology and availability. Recent developments in hybrid technology may shrink these numbers by a factor of better than 5:1 but costs will probably remain about the same.

These results are summarized in Table 4.10 and 4.11. Table 4.11 lists the power requirements and number of modules that were estimated for the complementary missions due to sensor similarity with the basic missions.

b. Assuming an arbitrary wattage and an estimated number of interfaces for each power module, the number of modules required for each sensor was estimated by counting the total number of power interfaces for each sensor as shown on the GO diagrams.

The quantity representing each different type of power module was then multiplied by the number of flights projected for that mission during the 1981-1985 time period. This is best seen in Table 4.12 which lists the number of power modules needed for all flights by mission year, the number of spares and the total procured. The number of spares which are required is heavily dependent upon the sensor development schedule since appropriate SIM or simulators will have to be supplied to the sensor manufacturers. A preliminary estimation of spares is presented in Table 4.13.

Table 4.14 itemizes several of the basic physical characteristics for each of the four power modules.

4.4.2 Standard Data Handling Systems

The number of Data Handling Units (DHU's) and Master Data Sequencer and Controllers (MDSC's) required for each typical mission was determined as follows. Only sensors having expected data bit rates greater than 64 Kbps were considered to be valid candidates. Of these, one simple DHU was assigned to each sensor having expected data bit rates less than 1 Mbps. For data rates greater than 1 Mbps, it was assumed that a complex DHU would be required. It was assumed that one simple MDSC would be required for each mission flight using simple DHU's, and one complex

Table 4.10 Total System Power Estimates-Basic Missions

System	Number of Modules				**INT	Power		Number of Model Components
	LPS	RPS	HPS	PPS/UPS		Avg	Peak	
STORMSAT	1	9	-	1		156	225	569
LANDSAT	1	10	2	1		207	300	836
TIROS	1	5	4	1	(3)	152	270	1[illegible]56
HEATE-1	1	3	5	1		94	130	1737
HEATE-2	1	3	5	1		91	125	943
GRE	1	3	5	1		87	125	1367
*SEASAT	1	7	5	4	(7)	358	1400	-
							Total:	6708

*SEASAT...Estimates are from total power required for each sensor. SEASAT includes 5 active radiating sensors plus 2 passive sensors.

**INT.... Indicates these missions are presently designed to carry dedicated power supplies for certain INTernal sensors. The number of sensors, or individual power supplies involved are given in this column.

Table 4.11 Estimated Power Requirements - Complementary Missions

System (Power Ref. Sys)	Number of Modules LPS	RPS	HPS	PPS/UPS	Estimate Power Avg (Watts)
SOLARMAX (TIROS)	1	5	4	1	150
SPACE TELESCOPE (TIROS)	1	5	4	1	150
HEAO-BLK2 (TIROS)	1	5	4	1	150
ENVIR MON (TIROS)	1	5	4	1	150
EARTH SURVEY (LANDSAT)	1	10	2	1	207
SEOS (LANDSAT)	1	10	2	1	207
SOLAR OBSER. (TIROS)	1	5	4	1	105

Table 4.12 Power SIM Quantities

SIM	Mission Year 1981	82	83	84	85	Flt	Spares	Total
LPS	4	5	7	7	8	31	48	79
RPS	27	37	49	53	57	223	88	311
HVS	11	17	20	16	27	91	50	141
PPS/UPS	4	8	7	7	11	37	48	85

Table 4.13 Preliminary Estimation of Spare SIM

Mission	No. of Sensors Requiring SIM/No. of SIM Required on Flight							
	LPS	RPS	HVS	PPS/UPS	DHU-S	DHU-C	MDSC-S	MDSC-C
STORMSAT	2/1	2/8	-	2/1	2/4	-	2/2	-
LANDSAT	2/1	2/10	1/2	2/1	-	2/4	-	2/2
TIROS	4/1	3/5	1/4	1/3	-	-	-	-
HEATE II	4/1	4/3	3/4	4/1	-	-	-	-
SEASAT	6/1	6/7	6/5	6/4	1/2	-	1/2	-
Solar Max.	9/1	9/5	8/4	9/1	-	-	-	-
Spare Tele.	3/1	3/5	3/5	3/1	3/6	-	3/2	-
HEAD BLK II	8/1	8/5	8/5	8/1	-	-	-	-
Env. Mon.	6/1	6/5	6/5	6/1	1/2	-	1/2	-
Earth Surv.	2/1	2/10	2/2	2/1	-	2/4	-	2/2
SEOS	8/1	8/10	2/2	8/1	4/8	-	4/2	-
Solar Max.	8/1	8/5	8/4	8/1	-	-	-	-
No. of SIM Required If Supplied to Each Sensor Manufacturer[(1)]	62	90	55	61	22	8	13	4
Average Number of Spares								
• Flight (10%)	4	23	10	4	6	3	4	2
• Ground	37[(2)]	54[(2)]	33[(2)]	37[(2)]	12[(3)]	6[(3)]	4[(3)]	3[(3)]
• 20% Ground	7	11	7	7	3	2	1	1
Total Spares	48	88	50	48	21	11	9	6

[(1)] Summation of the larger of XX/YY.

[(2)] Based upon average of three-year supply.

[(3)] Based upon specific mission timing.

MDSC would be required for each mission flight using complex DHU's. It was also assumed that each mission flight would carry an operational data handling system and a commandable standby data handling system.

Tables 4.15 and 4.16 summarize the number of operational and standby components needed for a single flight on each mission. Table 4.17 depicts the quantities of each Data Handling component needed by year from 1981 through 1985, with reasonable allowance for flight and ground spares. The spares are as determined in Table 4.13. For comparison, Tables 4.18 and 4.19 have been prepared to show the number of simple DHU's and partial DHU's needed if channel division and multiple sample and hold techniques are used in place of the complex DHU's and MDSC's on those missions which have a projected data bit rate greater than 1 Mbps.

Table 4.14 SIM Characteristics

Power Modules	Weight (kg)	Volume (cm3)	Power (Watts)
Logic Power Supply	1.23	850	25
Regulated Power Supply	1.23	850	25
High Voltage Power Supply	1.45	991	2
Pulsed & Unreg. Power Supply	4.09	2832	100
Power Pack Efficiency:	70%		

Table 4.15 Data Handling Unit Estimates - Basic Missions

System	Number of Modules (Oper + Standby)				Max Bit Rate
	DHU* (S)	DHU* (C)	MDSC* (S)	MDSC* (C)	
STORMSAT	2+2		1+1		**10.5 Mbps, 1 Kbps
LANDSAT		2+2		1+1	15 Mpbs, 115 Mbps.
TIROS	1+1		1+1		0.67 Mbps.
HEATE-1	-	-	-	-	5.8 Kbps.
HEATE-2	-	-	-	-	2 Kbps.
GRE	-	-	-	-	4 Kbps.
SEASAT	1+1		1+1		85 Kbps, 22.6 Kbps.

**It has been assumed that the bit rate on STORMSAT can be lowered to 1.0 Mbps or lower due to the geosynchronous orbit and hence a slower rate of change in the data is expected; otherwise, one complex DHU or a number of simple DHU's will be required.

*S = simple DHU, or MDSC
C = complex DHU, or MDSC

Table 4.16 Data Handling Unit Estimates - Complementary Missions

System (Data Ref. Sys)	Number of Modules DHU*(S)	DHU*(C)	MDSC*(S)	MDSC*(C)	Max Bit Rate
SOLAR MAX (HEATE-1)	-	-	-	-	<6 Kbps.
SPACE TELES (TIROS/HDR)	3+3		1+1		0.67 Mbps.
HEAO-BLK2 (HEATE-1)	-	-	-	-	<6 Kbps.
ENVIR MON (TIROS/HDR)	2+2		1+1		0.67 Mbps
EARTH SURV (LANDSAT)		2+2		1+1	115 Mpbs.
SEOS (STORMSAT)	**4+4		1+1		10.5 Mbps
SOLAR OBS (TIROS/HDR)	-	-	-	-	0.67 Mbps.

**Assumed bit rate for SEOS can be lowered to 1.0 Mbps or lower.

*S = simple DHU, or MDSC
C = Complex DHU, or MDSC

Table 4.17 Data Handling SIM Quantities

SIM	Mission Year 1981	82	83	84	85	Flt	Spares	Total
DHU*(S)	6	8	14	14	10	52	21	73
MDSC*(S)	4	6	8	8	4	30	11	41
DHU*(C)	4	8	8	8	8	36	9	45
MDSC*(C)	2	4	4	4	4	18	6	24

*S = simple DHU, or MDSC
C = complex DHU, or MDSC

Table 4.18 Additional Data Handling Units - By Mission (Units per mission flight assuming no complex units available)

Mission	[1]Partial DHU(1)	[2]Partial DHU(2)	[3]MDSC(S) (Opr. + Standby)
STORMSAT (AASIR)	3	9	1+1
LANDSAT (MMS)	4	16	1+1
(TMS)	24	96	
EARTH SURV	24	96	1+1
SEOS	3	9	1+1

[1]Partial DHU(1).. A unit in which the Analog Mux Sections are deliberately left inoperative.

[2]partial DHU(2).. A unit in which the Sample + Hold, & ADC sections are deliberately left inoperative.

[3]MDSC must be hardwired to handle data rates greater than 1 Mbps.

NOTE: No allowance has been made for standby partial DHU's.

Table 4.19 Data Handling Units (Assuming No Complex Units Available)

SIM	1981	82	83	84	85	Flt	Spares	Grand Total
COMPLETE DHU	6	8	14	14	10	52	21	73
[1]PARTIAL DHU(1)	121	224	217	242	233	1037	443*	1480
[2]PARTIAL DHU(2)	31	56	55	62	59	263	114*	377
TOTALS:	158	288	286	318	302	1352	578	1930

[1]Partial DHU(1): A unit in which the Analog Mux sections are deliberately left inoperative.

[2]Partial DHU(2): A unit in which the Sample & Hold and ADC sections are deliberately left inoperative.

NOTE: No allowance has been made for standby partial DHU's.

*Rough estimates.

5. ECONOMIC ANALYSIS

5.1 Methodology

The goal of standardization can be expressed as the reduction of the present value of the cost of performing a specified mission model. Standardization has the potential of reducing the present value of the costs through reductions in nonrecurring (design, development, test, engineering and related investments) costs resulting from the need to develop fewer different subsystems than would otherwise be necessary if specialized sensor subsystems were utilized (development costs foregone). Costs may also be reduced through larger "buys" and associated learning effects. On the other hand, costs tend to increase because standardized subsystems will not match sensor requirements as closely as specialized sensor subsystems; for example, a finite degree of subsystem modularization will tend to yield spacecraft which, in general, have more capability than some missions require. This additional capability tends to increase unit recurring costs and transportation related costs.

There are other costs associated with standardization which are associated with the carrying of inventory or spares. For example, consider the case where there are n sensors which are to be placed on board a specific MMS flight and each requires power which will be supplied by a logic power standardized interface module. The experimenter requires the power for the logic circuits in order to check out and qualify his sensor prior to placing the sensor on the spacecraft. This power can be provided by either a simulator or an actual logic power standardized interface module. If the logic power SIM is capable of providing power for all n sensors then an additional cost--

either for power SIM simulators or for n-1 SIMs*--must be charged to the SIM alternative.

The basic task of economic analysis is to determine that alternative which minimizes the present value of the cost of performing the mission model. There are two basic alternatives to be considered, namely, (a) the development and use of specialized sensor subsystems (i.e., business as usual) and (b) the development and use of standardized interface modules which may be used in lieu of the specialized sensor subsystems. Actually, there are many more alternatives since different levels of SIM modularity may be considered as well as mixed strategies (across the mission model and within a spacecraft) of utilization of both SIM and specialized snesor subsystems. Each of these also needs to be considered in order to determine the alternative which minimizes the present value of the cost of performing the mission model. (Section 8.3 describes a general economic analysis methodology which utilizes an integer programming approach to determine the specific alternative--including mixed strategies--which minimizes the present value of the cost of performing the mission model.)

The objective of the current economic analysis is to evaluate the economic benefits which may result from the development and utilization of one or more standard interface modules and, as a result, develop insights into which standardized interface modules should be developed. The economic benefits are measured in terms of one alternative relative to another alternative. It is convenient to establish a base-line alternative with which all other alternatives can be compared. For the case at hand, the base-line

*Actually, somewhat less than n-1 need be charged because some of these will in reality be utilized on following missions.

alternative is that of pursuing business as usual, i.e., specialized sensor subsystems. The annual benefits are, therefore, measured as annual savings of an alternative involving standardization measured relative to the base-line costs. This is illustrated in Figures 5.1 and 5.2 where A and B represent two alternative courses of action with A being the base-line alternative. Figure 5.1 illustrates the annual cost of performing the mission model with these two alternatives each having different cost streams over the time period considered. The differences in the cost patterns are due, among other things, to differences in (a) the number of subsystems requiring development, (b) the timing of subsystem developments, (c) the number of subsystems required as a function of time and (d) the unit recurring costs.

Figure 5.2 illustrates the annual cost savings (i.e., annual benefits) of alternative B relative to alternative A. These cost savings represent the shaded area in Figure 5.1. The question arises as to whether or not it is desirable to pursue alternative B rather than alternative A. In the absence of annual budget constraints, the decision as to which alternative to pursue can be made based upon the net present value of the annual savings (loss) as depicted in Figure 5.2.

The computation of net present value (NPV) seeks to adjust cash flows (the stream of savings and loss) occurring in different future time periods in a manner so as to eliminate time as a parameter. The adjustment process, known as discounting, establishes a present or "now" value of the future cash flows. The rationale behind the adjustment is that a dollar received in the future is worth less than a dollar today, since the dollar in hand today could be used to improve one's status today rather than at some

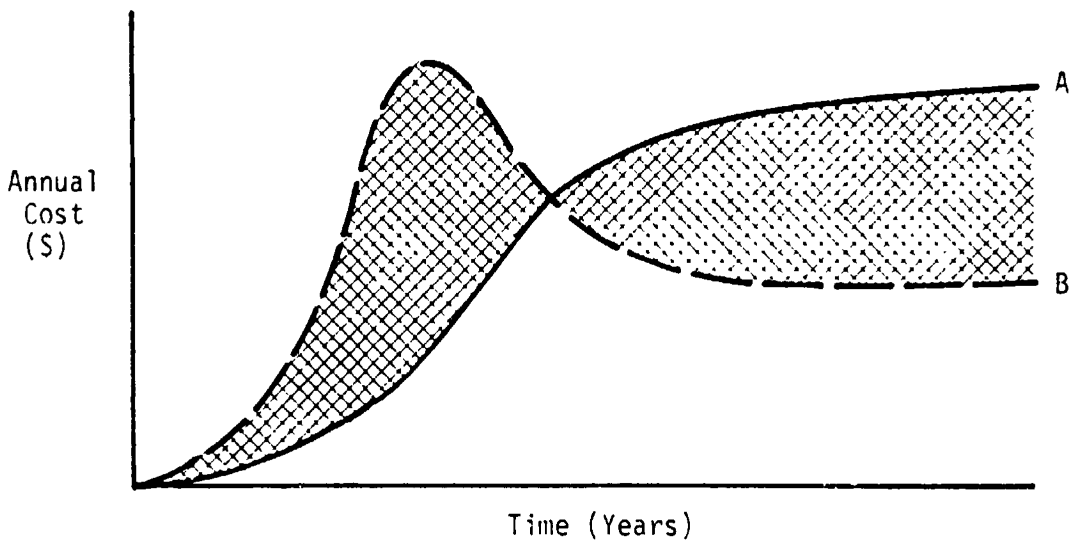


Figure 5.1 Annual Cost of Alternatives A nd B

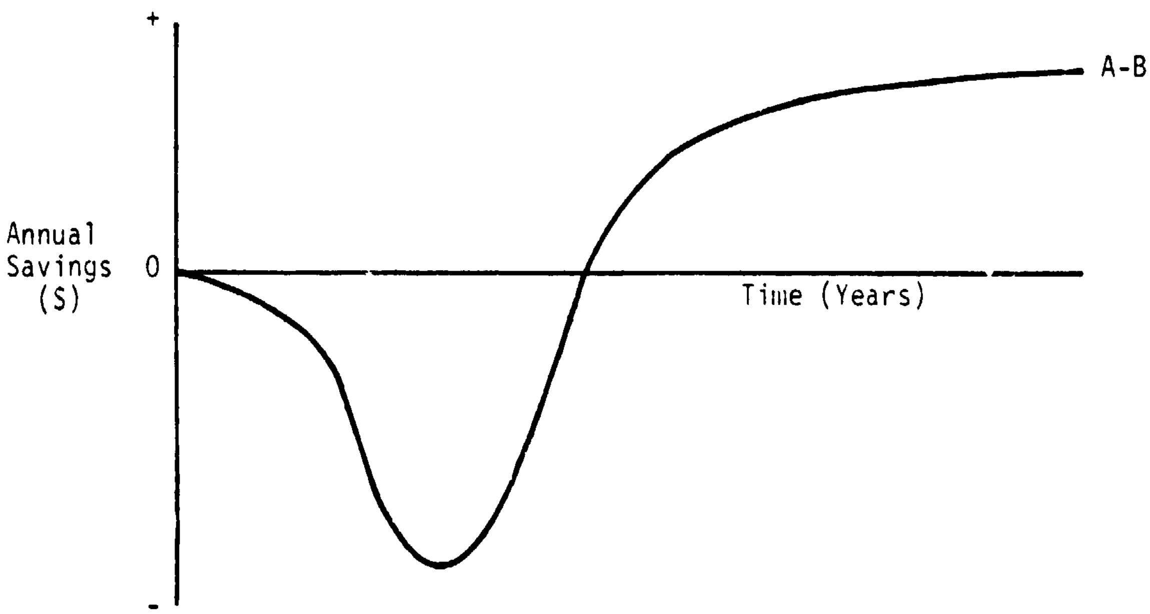


Figure 5.2 Annual Savings of Alternative B Relative to Alternative A

point in the future. The computational mechanism is to reduce the cash flow occurring in a particular future period by a discount factor such that the discounted amount is the amount which, if invested at the discount rate from the present to the corresponding future time, would be equal to the unadjusted value. In the mathematical sense, this process is the complement of compounded interest on a savings account; although, in the economic sense, discounting is a very different concept. The net present value computation is as follows:

$$NPV = \sum_{i=1}^{N} CF_i/(1 + r/100)^i$$

where CF_i represents the cash flow in the i^{th} time period, N is the planning horizon and r is the discount rate (percent) or cost of capital. An interpretation of the net present value of an investment is that it represents the maximum sum of money that an investor with an adjusted (for inflation) cost of capital equal to r would be willing to pay so as to have the opportunity to invest. It represents the value of the project over and above all costs associated with funding the project (in the private sector this includes interest expenses paid at the cost of capital equal to r). A positive NPV indicates a return in excess of the project cost plus the cost of capital. In theory, all projects having $NPV > 0$ should be undertaken. Those projects with $NPV < 0$ should not be undertaken, and for those projects with $NPV = 0$, the choice is immaterial. The previous statements, of course, are true in a world of certainty.

Central to the use of the NPV criteria is the choice of the appropriate discount rate. Although economists have generally agreed that the adjusted weighted average cost of capital is the appropriate rate to be used for the private sector, much controversy still exists regarding the appropriate rate for government use in decision making. Some have maintained that the long-term government bond rates are the most appropriate. Others have maintained that the rate should be no lower than the typical rate of return achieved by investments in the private sector. OMB has set a rate of 10 percent for use in evaluating government projects on an equitable basis. The OMB discount rate, unless otherwise noted, will be used in all present value computations.

Since the goal of this current study is to establish preliminary estimates of the economic benefits of developing standardized interface modules and to indicate those standardized interface modules which deserve further detailed considerations, a number of simplifying assumptions have been made. In particular, it is assumed that costs are incurred instantaneously (i.e., no cost spreading), differences in transportation charges resulting from differences in mass and/or volume are not considered, differences in probability of mission success caused by the use of standard interface modules are not considered (with the exception of identifying those SIM components which are critical from a reliability standpoint and hence providing redundancy), and it is assumed that the SIM will be utilized on all sensors, as indicated by the mission model of Figure 3.1, with unity probability. Further, an attempt was made to include the cost of spares and ground support equipment in the form of additional SIM or simulators for the sensor developers. It should be cautioned that a detailed analysis of the sparing and ground support equipment requirements was not performed as part of this study.

Within the context of the above simplifications, the present value of the cost of the business as usual alternative, NPV_s, and the standardized interface module alternative, NPV_s', are obtained from

$$NPV_s = \sum_{i=1}^{N} COST_{i,s}/(1 + r/100)^i$$

$$NPV'_s = \sum_{i=1}^{N} COST'_{i,s}/(1 + r/100)^i$$

$$\Delta NPV_s = NPV_s - NPV'_s$$

where COST represents annual cost, the primed quantities represent the SIM alternative and ΔNPV_s is the economic value of developing and using the s SIM. It should be noted that the annual cost associated with the business as usual approach is subscripted with s indicating that $COST_{i,s}$ is the annual cost of the business as usual approach which will be impacted by the development of the s SIM.

The annual cost of the business as usual approach is given by

$$COST_{i,s} = \sum_{f=1}^{F_i} \left\{ \sum_{e=1}^{E_f} (NRC_{e,f,s} + RC_{e,f,s}) \right\}$$

where, as stated previously, $COST_{i,s}$, is the cost associated with those subsystems which can be removed from the various sensors if the s SIM is used. Note that the cost is not that of the complete sensor but only that portion of the sensor which will be replaced by the standardized interface module. f is an index associated with flights,* F_i is the number of flights in year i, e is an index associated with sensors or experiments, E_f is the number of experiments on flight f, $NRC_{e,f,s}$ is the nonrecurring cost associated with those functions in the e^{th} experiment

*Note that the mission concept is not considered explicitly. What is considered is a sensor model which describes which sensors are launched each year.

which can be replaced by the s SIM, and $RC_{e,f,s}$ is the recurring cost of those functions in the e^{th} experiment which can be replaced by the s SIM. It should be noted that if the same or similar sensor or experiment was developed previously, then NRC may be zero or a function of the original nonrecurring cost and RC may be reduced by learning effects.

The annual cost of the alternative based upon the use of standardized interface modules is given by

$$COST'_{i,s} = NRC'_{i,s} + \sum_{f=1}^{F_i} NSIM_{f,s} * RC'_f$$

where the first term is the nonrecurring cost associated with the s SIM and the second term is the summation of the SIM recurring cost per flight with $NSIM_{f,s}$ being the number of SIMs required for the f flight.

This general economic analysis methodology is illustrated in Figure 5.3. The basic inputs to the analyses are the specification of the mission model in terms of the specific sensors which are to be utilized as a function of time, the configuration of the sensor subsystems which may be impacted by the standardized interface modules, and the characteristics and configuration of the standardized interface modules. The specification of the sensors as a function of time provides the basis for the time distribution of costs and the determination of present value of costs, the specification of the SIM together with the sensor subsystem characteristics allows the applicability of the SIM to the specific sensors to be established and hence, using the PRICE costing methodology (to be described in following paragraphs), allows the cost of the SIM and the applicable sensor subsystem costs to be established.

As will be discussed in Section 5.2, standardized interface module specifications have been developed to a

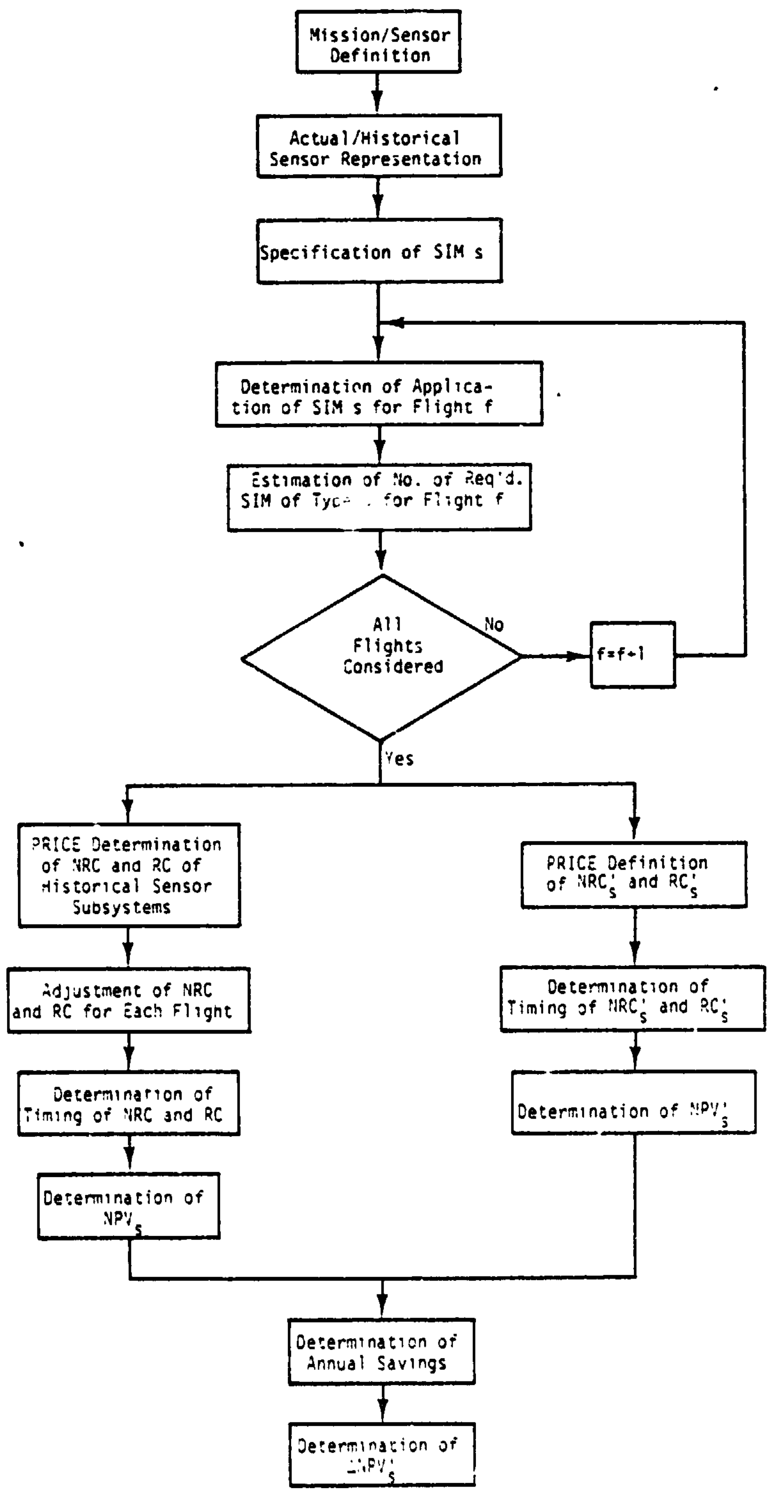


Figure 5.3 Basic Economic Analysis Methodology

sufficient level of detail so that preliminary estimates of nonrecurring and recurring costs are possible. Unfortunately, detailed definitions (at the subsystem level) of sensors for missions in the 1980-85 time period were not available. Since the economic value of standardized interface modules can only be determined by comparing the costs (and present values) of the SIM alternative relative to the costs (and present value) of the business as usual alternative, it is necessary to establish the costs of the sensor subsystems which would be impacted by the SIM. Because of the lack of definition this is not possible. Therefore it was decided to relate the future (or actual) sensors to historical sensors which it is anticipated would have the same, or very similar, subsystems (see Tables 3.16 and 3.17). The historical sensors provide the detailed definition at the subsystem level such that the PRICE methodology can be used to estimate both nonrecurring and unit recurring costs. The historical sensor subsystem cost estimates can then be used to establish cost estimates for the future sensor subsystems by making subjective estimates pertaining to anticipated level of relative complexity. Subjective estimates of learning effects for both the recurring and nonrecurring costs can be made for the sensors for each flight in terms of the specific time of use and previous developments. The buildup of the annual costs (for the business as usual alternative) is illustrated in Figure 5.4 based upon the specific subsystems in each sensor which are impacted by the s SIM.

Determination of the economic benefit derived from the standardization of common functions requires the hardware to perform the functions to be identified and characterized, and a cost estimation procedure is required to accurately determine hardware costs. The cost estimation

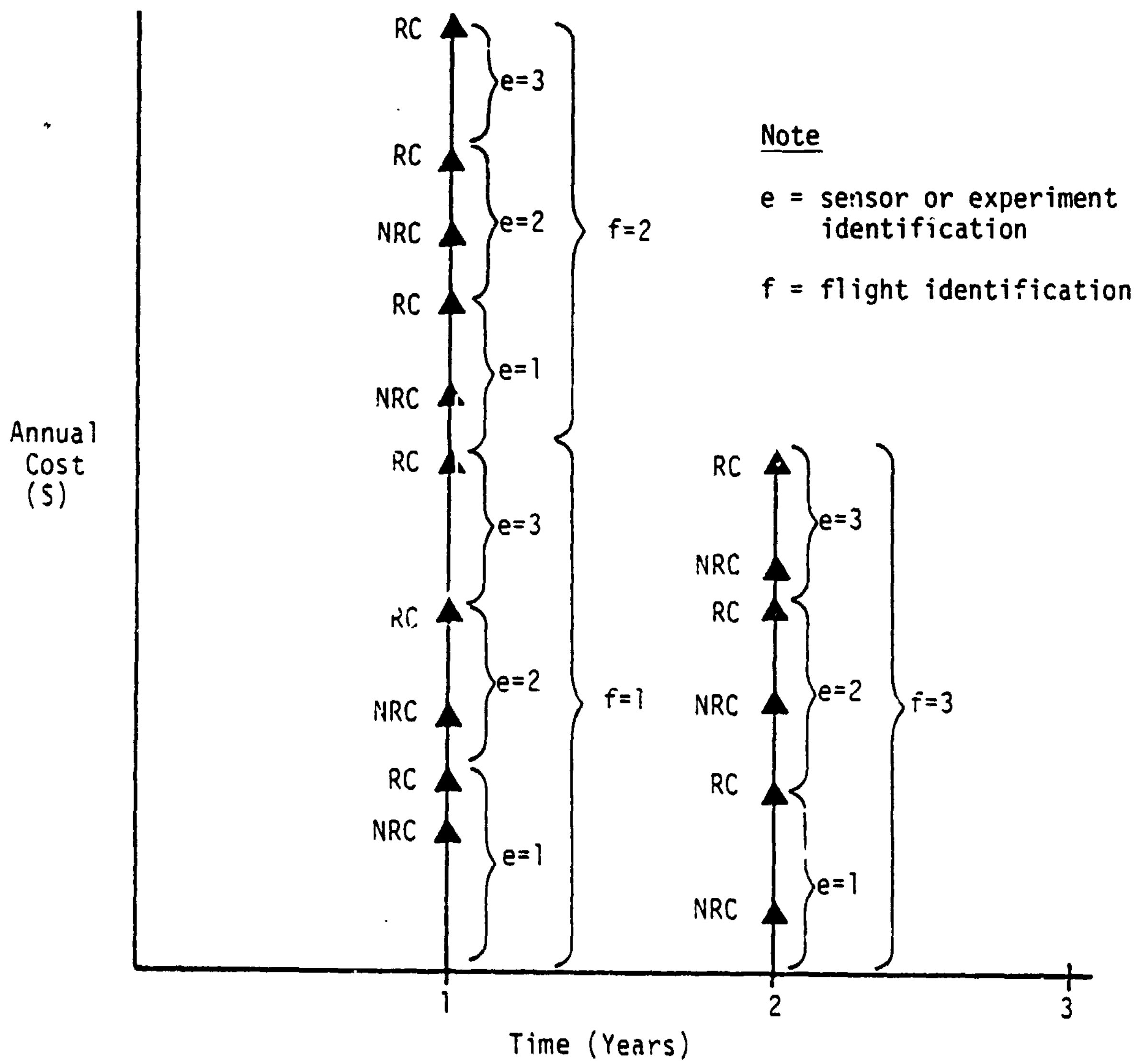


Figure 5.4 Annual Cost Buildup in Terms of Sensor Subsystems Impacted by a Particular SIM (Business as Usual Alternatives)

procedure to be used must be applicable when only preliminary hardware information is available. There are several cost modeling procedures which can estimate hardware costs based upon similarities and extrapolation of previous cost histories. The "PRICE" cost estimating methodology was used in this study and is a computerized technique developed by RCA Corporation over a 13-year period. The program uses simple descriptions of the hardware physical characteristics, complexities and state of development to obtain a cost estimate for development and production costs. The model has been extensively used throughout industry with good results for a broad category of hardware.

The PRICE program is based upon a computerized mathematical model which can determine hardware development and production costs based upon a relatively simple description of hardware and program elements. The algorithms used are based upon a large collection of historical data and are applicable for a wide assortment of equipment types.

The input data for the program describes the fundamental characteristics upon which cost is most dependent; namely, the complexity of the hardware and the quantity of hardware of that complexity that is to be developed and produced. Additional information is also provided and describes previous experience with development of similar hardware, environmental considerations and engineering and production schedules, all of which are combined and interrelated to assess the total costs.

The PRICE program determines and lists recurring and nonrecurring costs for the development and production phases of the described hardware. The costs are developed for engineering during the development phase and for the production phase, and manufacturing costs are derived for the development phase to support engineering effort to develop models and prototypes and for the production phase. The PRICE methodology is highly interactive, in that the hardware physical descriptions, complexities and schedules, which are strong determinants of hardware costs, can be easily varied and resulting cost variations noted. If the input data describes a complex device with a difficult short schedule, or a nonoptimum overlap of engineering and production schedules, the program costs will be affected accordingly; an extreme case will be automatically error flagged by the program. Similarly, if the physical description of the hardware does not fit a historical pattern of mechanical and electrical densities for the category of hardware, the program will error flag the inconsistencies. The physical, historical and schedule descriptors must be made to fit the concept of the product in order to achieve reasonable cost accuracies.

The PRICE methodology basically derives product costs using preliminary conceptual descriptors, as opposed to detailed parts counts and task estimates. In addition, the methodology accounts for:

- changes in engineering and manufacturing technology developments during the life of the program,
- effect of economic adjustments,
- normal, accelerated and protracted schedules,
- stoppage and restarting a production run,
- design redundancy within the hardware, and
- reliability and testing requirements.

The PRICE program, due to its simplicity of conceptual product descriptions, and the parametric nature of the cost estimation procedure provides an ideal method to assess the economic benefits of the functional standardization through the use of standardized interface modules and the Multi-Mission Spacecraft. A full description of the input data and definitions is given in the appendix, Section 8.1.

5.2 Cost Estimation

The PRICE program was used to develop cost estimates for the standardized interface modules as detailed in Section 4.0 and for the sensor subsystems as described in Section 3.0, which will be impacted by the use of the standard interface modules. This section is concerned with summarizing the cost estimation procedures, assumptions, PRICE input data and computed costs.

The standard interface modules defined from the analysis of the requirements (Section 4.0) were characterized to the extent required for PRICE input data. Typically, this characterization process required an estimate of the weight, volume and power of the standard module, and a determination of the type of circuit elements expected for the device. The mechanical and electrical densities determined from the estimates were compared with historical data to verify the correctness of the estimates. Any variations from typical values led to corrections of the input data, or justification of rationale of the original estimates. To determine the historical background of the standard module, it was assumed that the hardware would be procured from qualified contractors with experience in the particular area of expertise required, and no "state-of-the-art" development would be required. To determine the engineering and production schedules for the standard modules, it was assumed that procurement for the devices would start sufficiently in advance of the need for the hardware to allow a "normal" engineering and manufacturing effort. The manufacturing effort was assumed to be for the total quantity required, to maximize the effects of the economy of a large, unbroken production run. All costs were computed with a 30 percent markup from direct engineering costs to account for IR&D, G&A and fees. The specific assumptions for the required PRICE inputs are described in the following paragraphs.

5.2.1 Regulated Power Supply (RPS) SIM

The standard interface module considered for the regulated power supply function consists of a redundant power supply, whose active circuitry is selectable by command. Based upon existing technology, the physical characteristics were estimated as indicated in Table 5.1. The mechanical portion of the RPS weight is a relatively high percentage of the total weight, due to the expected use of shielding, transformers and relay. The circuitry was characterized as typical of a pulse width modulator (PWM) type power regulator.

The RPS historical and complexity descriptions are considered to characterize a device which is developed from a modification of a similar design, using techniques within the existing state-of-the-art. Only half of the defined quantity of electronics requires design effort with the remainder of the hardware being redundant. The resulting mechanical and electrical design is considered to have a manufacturing compexity typical of an "average" space qualified power supply. The estimated historical and complexity characteristics are summarized in Table 5.1 along with the assumed program schedule and quantities procured.

The procurement philosophy assumes a 28-month program such that the bulk of the required RPS units are produced in time for integration and test activity one year prior to the first launch, with the remainder of the units at a continuous rate reasonable to the estimated manufacturing complexity. Two prototype units were considered, to debug the engineering and production problems. Typical NASA program reliability and documentation requirements were assumed. The flight quantity was derived from the analysis of requirements discussed in Section 4.0.

5.2.2 Logic Power Supply (LPS) SIM

The logic power supply SIM characterization is identical to that of the RPS, differing only in the quantities

Table 5.1 Regulated Power Supply (RPS) SIM Characteristics	
Physical Characteristics	
Total Unit Weight	1.23 kg
Total Unit Volume	850 cm^3
Total Unit Power	7.5 watts
Mechanical Weight	.12 kg
Packaging Density	70% full
Circuitry	Mixed digital, analog
Complexity Characteristics	
Mechanical Producibility	Typical Average
Electrical Producibility	Typical Average
Mechanical Design Repeat	20%
Electrical Design Repeat	50%
New Mechanical Design	50%
New Electrical Design	15%
Engineering Heritage	Routine Modification
Program Characteristics	
Engineering Development Time	3 months to release
Production Time	24 months
Quantity of Flight Units*	369
Prototypes	2
Year of Procurement Start	1978
Programmatic Requirements	
Systems	"Average"
Data and Documentation	"Average"
Tooling and Test Equipment	"Average"
Markup from Direct Costs	1.3

*LPS and RPS are identical except for the number of flights with the LPS flights being equal to 59.

required for each unit. This rationale is based upon the similarity of the two units in terms of physical characteristics, complexities, design history and procurement philosophy. For the purposes of the economic analysis, the LPS is assumed to be procured on a separate contractual relationship; however, further economies may be realized if the RPS and LPS were procured as two variations of a single hardware "buy", as the RPS and LPS differ only in the range of output voltages required.

5.2.3 High Voltage Supply (HVS) SIM

The HVS is characterized as a redundant programmable multiple high voltage source housed in a common package. As in the case of the RPS and LPS, the HVS is assumed to be procured from a qualified, knowledgeable contractor with a schedule optimum for development and production. Physical characteristics assumed are shown in Table 5.2. The HVS physical characteristics are similar to the RPS and LPS, however, a higher packaging density typical of a high voltage power supply is assumed with an attendant decrease in percentage of mechanical weight. The HVS complexity characteristics are also shown in Table 5.2. The characterization of the HVS complexity is influenced by the increased density, and general increased design and manufacturing complexity, of a high voltage source. Care is generally required in lead routing, length, terminations and proximity to other wires. Design techniques to minimize effects of corona and arcing in a vacuum environment are generally known, but result in a hardware design more complex to produce.

The assumed procurement philosophy is summarized in Table 5.2 for the HVS and is expected to be longer than the simpler RPS and LPS designs due to the increased engineering content of the design. The manufacturing cycle is completed in approximately the same time scale as for the simpler RPS. Although the unit is more complex to produce,

Table 5.2 High Voltage Supply (HVS) SIM Characteristics

Physical Characteristics	
Total Unit Weight	1.45 kg
Total Unit Volume	991 cm^3
Total Unit Power	2 watts
Mechanical Weight	.18 kg
Packaging Density	85% full
Circuitry	Mixed digital, analog
Complexity Characteristics	
Mechanical Producibility	More complex than "average"
Electrical Producibility	More complex than "average"
Mechanical Design Repeat	20%
Electrical Design Repeat	50%
New Mechanical Design	50%
New Electrical Design	15%
Engineering Heritage	Routine Modification
Program Characteristics	
Engineering Development Time	8 months to hardware release
Production Time	24 months
Quantity of Flight Units	162
Prototypes	2
Year of Procurement Start	1978
Programmatic Requirements	
Systems	
Data and Documentation	Average
Tooling and Test Equipment	
Markup from Direct Costs	1.3

fewer HVS units are required, resulting in a lower production rate.

5.2.4 Pulse Power Supply (PPS/UPS) SIM

The pulse power supply module is characterized as a dual redundant unit, containing mixed analog and digital circuitry, with several large components associated with the isolator. The procurement assumptions are identical to those made for the RPS, LPS and HVS, namely, no state-of-art change, extrapolation from existing designs, adequate delivery schedule and procurement during 1978-1980 time period. The unit physical, complexity and program characteristics are given in Table 5.3.

5.2.5 Data Handling Unit (DHU) SIM

The data handling unit modules were characterized in two variations, one with a low data rate capability, and one with a high data rate capability. The low data rate DHU would be utilized for applications outside of the MMS data rate capability; the high data rate unit would be used for the very high data rates expected for some of the proposed experiments. The hardware for both modules has similar physical and program descriptions, however, the high data rate module will require an extensive development effort, with areas of new design to be considered. The costs associated with the additional developmental effort envisioned for the high data rate unit are obtained by appropriate adjustment of the historical background and amount of new electrical design anticipated for the module. The final design configuration is assumed to contain more advanced component technology. The resulting hardware would have a high bit rate capability but would not differ greatly in physical characteristics from the slower DHU modules. The characterization of the two DHU variations is shown in Table 5.4.

Table 5.3 Pulse Power Supply (PPS/UPS) SIM Characteristics

Physical Characteristics	
Total Unit Weight	4.1 kg
Total Unit Volume	2832 cm^3
Total Unit Power	30 watts
Mechanical Weight	.91 kg
Packaging Density	70% full
Circuitry	Mixed digital, analog
Complexity Characteristics	
Mechanical Producibility	Typical Average
Electrical Producibility	Typical Average
Mechanical Design Repeat	20%
Electrical Design Repeat	50%
New Mechanical Design	50%
New Electrical Design	5 %
Engineering Heritage	Simple design modification
Program Characteristics	
Engineering Development Time	6 months to hardware release
Production Time	24 months
Quantity of Flight Units	69
Prototypes	2
Year of Procurement Start	1978
Programmatic Requirements	
Systems	Average
Data and Documentation	Average
Tooling and Test Equipment	Average
Markup from Direct Costs	1.3

Table 5.4 Data Handling Unit (DHU) SIM Characteristics

	Low Data Rate Unit	High Data Rate Unit
Physical Characteristics		
Total Unit Weight	.68 kg	.68 kg
Total Unit Volume	651 cm^3	651 cm^3
Total Unit Power	0.2 watts	0.2 watts
Mechanical Weight	.07 kg	.07 kg
Packaging Density	85% full	85% full
Circuitry	Digital, mixed IC	Digital, mixed CMOS
Complexity Characteristics		
Mechanical Producibility	Typical Average	Typical Average
Electrical Producibility	Typical Average	Typical Average
Mechanical Design Repeat	20%	20%
Electrical Design Repeat	20%	20%
New Mechanical Design	50%	50%
New Electrical Design	20%	50%
Engineering Heritage	Simple Modification	Simple Modification
Program Characteristics		
Engineering Development Time	12 months	12 months
Production Time	24 months	24 months
Quantity of Flight Units	79	46
Prototypes	2	2
Year of Procurement Start	1978	1978
Programmatics		
Systems; Data and Documentation; Tooling and Test Equipment	Average	Average
Markup from Direct Costs	1.3	1.3

5.2.6 Master Data Sequencer and Controller (MDSC) SIM

The master data sequencer and controller modules have been modeled in two variations as was done with the DHU modules. The high data rate MDSC will be matched with high data rate DHU modules, and likewise, the low data rate MDSC modules will be employed with low data rate DHU modules. As in the case with the DHU modules, the MDSC modules are nonredundant and are used in pairs at the next assembly level to provide the redundancy required. The rationale for the physical complexity, and programmatic characteristics assumptions is similar to the considerations used to develop the characterization for the DHU modules. The MDSC characterization data is shown in Table 5.5.

5.2.7 Derivation of Inputs for Integral Designs

The cost savings achieved from the procurement of sensors which do not contain hardware for the functions performed by the standard interface modules is required in order to assess the economic benefit of interface module standardization. To avoid the uncertainty and inaccuracy involved in calculating total sensor cost inclusive of the standard function hardware (i.e., the hardware whose functions are replaced by the standard interface modules), and subtracting the total cost of the sensors without the standard function hardware, an approach was taken wherein the cost of the deleted function was obtained directly. To obtain these costs directly, the following assumptions were made:

- The design complexity of the hardware needed to perform a standard function may be simpler if included as part of a sensor, since the hardware is only required to service the specific requirements of the sensor and not multiple requirements of many sensors.

- The physical characterization of the hardware required to perform a function is similar if the hardware is included as part of the sensor or as a separate standard interface module.

Table 5.5 Master Data Sequencer and Controller (MDSC) SIM Characteristics

	Low Data Rate Unit	High Data Rate Unit
Physical Characteristics		
Total Unit Weight	.68 kg	.68 kg
Total Unit Volume	651 cm^3	651 cm^3
Total Unit Power	0.2 watts	0.2 watts
Mechanical Weight	.068 kg	.068 kg
Packaging Density	85% full	85% full
Circuitry	Digital, IC	Digital, mixed CMOS
Complexity Characteristics		
Mechanical Producibility	Typical Average	Typical Average
Electrical Producibility	Typical Average	Typical Average
Mechanical Design Repeat	20%	20%
Electrical Design Repeat	20%	20%
New Mechanical Design	50%	50%
New Electrical Design	20%	50%
Engineering Heritage	Simple Modification	Difficult Modification
Program Characteristics		
Engineering Development Time	12 months	12 months
Production Time	24 months	24 months
Quantity of Flight Units	45	24
Prototypes	2	2
Year of Procurement Start	1978	1978
Programmatics		
Systems; Data and Documentation; Tooling and Test Equipment	Average	Average
Markup from Direct Costs	1.3	1.3

- A sensor contractor will not have the same degree of familiarity of the hardware or design required to perform a standard function as would a contractor who specializes in a specific type of device.

- The nonrecurring cost of a standard function design will be borne by every sensor contract with an integral design.

- The per unit recurring cost of the hardware to perform a standard function will be greater for an integral sensor design since the production quantities will be less for each manufacturer.

With these general assumptions, the cost of the hardware deleted from the sensors can be obtained by making adjustments in the following three PRICE characterizations:

- Complexity/Experience Adjustment: Consists of re-estimating and adjusting the hardware complexity and historical background to characterize the hardware function as it appears as part of the sensor package.

- Quantity Adjustment: Consists of determining the revised quantities associated with procurement as part of the sensor package to establish recurring and nonrecurring costs based upon smaller quantities and multiple procurements.

- Physical Characteristic Adjustment: Consists of modifying the physical descriptions of the standard interface modules to derive the characterization of hardware to be deleted.

5.2.8 Complexity/Experience Adjustment

It is anticipated that there will be differences in the cost of performing a specific function with a standard interface module and with hardware included as part of a sensor, the reason being different equipment complexities and different contractor experiences. These differences must be taken into account when establishing the cost of performing the functions with hardware included as part of the sensor. This is accomplished through the PRICE characterization of the hardware obtained by estimating the differences in the historical background and familiarity of the sensor contractor and the standard interface module contractor, and estimating the reduction in complexity of the hardware when included as part of the sensor. As a first cut at the estimated historical and complexity differences, each standard interface module was assessed to determine the degree of complexity, both in manufacturing and in engineering, introduced by the necessity of providing for multiple requirements. For several of the standard modules, the complexity is very nearly the same; e.g., a logic power supply providing voltages for CMOS devices would differ only in the total power output, as the specific sensor need and multiple sensor needs will generally require the same number and degree of regulation for the output voltages. Conversely, the high voltage power supply configured for the standard interface module must provide multiple high voltages on difference output lines; a high voltage supply designed for a specific sensor and included as part of the sensor will probably be less complex. To quantify these assumed differences, scale factors were determined to relate the history and complexity of the integral sensor designs with the standard interface modules. The scale factors are normalized to the characterizations derived for the standard interface modules, and are shown in Table 5.6. The scale factors listed are estimates of the degree of reduction or degree of increase for the listed elements, this "degree of difference" is used to adjust the PRICE coefficient which characterizes that particular area.

The data in Table 5.6 indicates that the design costs of an integral high voltage power supply (HPS) will reflect 25 percent of the engineering complexity, 50 percent of the manufacturing complexity and about the same design experience as a contractor who can provide the function in a standard interface module. A design experience scale factor greater than 1.0 indicates an increased cost due to reduced historical background in the design area.

Table 5.6 Normalized Historical and Complexity Scale Factors for Integral Design Relative to SIM Design

Unit	Relative Engineering Complexity	Relative Manufacturing Complexity	Design Experience
LPS	.9	1.0	1.5
RPS	.7	1.0	1.5
HPS	.25	.5	1.0
PPS/UPS	.6	.7	.8
DHU-S*	1.0	1.0	1.2
MDSC-S*	1.0	1.0	1.2
DHU-C*	.8	.8	1.0
MDSC-C*	.8	.8	1.0

*S = simple or low data rate.
C = complex or high data rate.

5.2.9 Quantity Adjustment

The sensor complements for each defined mission were examined to determine the number of times a nonrecurring cost would be expended (e.g., each sensor on a specific mission requiring an RPS would incur a nonrecurring cost for development of its unique RPS) and to determine the repetitive utilization of the integral hardware for assessment of recurring costs. It was assumed that the non-recurring costs and the recurring costs for a particular hardware function would be approximately the same for any sensor contractor.

Table 5.7 lists the missions considered, the sensor complements and the distribution of standard interface modules required for the sensors as derived in Section 4.0 of this report. Table 5.8 lists for each mission the cost

Table 5.7 Standard Interface Module Utilization by Mission

Sensor	LPS	RPS	HVS	PPS/UPS	DHU-S	DHU-C	MDSC-S	MDSC-C
STORMSAT								
AASIR	1	5	-	1	2	-	2	-
MASR		4	-		2	-		-
LANDSAT								
TMS	1	5	-	1	-	2	-	2
MSS		5	2		-	2	-	
TIROS								
AVHRR		1			2	-	2	-
TVOS								
BSU		-			-	-	-	-
SSU		-		1	-	-	-	-
MSU	1	-			-	-	-	-
SEM								
TED		1	3		-	-	-	-
MEPED		1	-		-	-	-	-
HEPED		1	1		-	-	-	-
DCS		1	-		-	-	-	-
HEATE 1								
XRS	1	2	3	1	-	-	-	-
XRC		1	2		-	-	-	-
HEATE 2								
GSA		1	1		-	-	-	-
CPA	1	1	-		-	-	-	-
GVE			1	1	-	-	-	-
GPE		1	2		-	-	-	-
GRE								
SCC		1/2	2		-	-	-	-
TASC	1	1	1	1	-	-	-	-
ACD		1/2	1		-	-	-	-
TOFC		1	1		-	-	-	-
SEASAT								
A		2	1	1	-	-	-	-
B		1	1/2	1/2	-	-	-	-
C	1	1	1	1	2	-	2	-
D		1	1	1/2	-	-	-	-
E		1	1	1/2	-	-	-	-
F		1	1/2	1/2	-	-	-	-
SOLAR MAX								
A		1/4	1/2		-	-	-	-
B		1/4	1/2		-	-	-	-
C		1/2	-		-	-	-	-
D		1/2	1/2		-	-	-	-
E	1	1/2	1/2	1	-	-	-	-
F		1/2	1/2		-	-	-	-
G		1/2	1/2		-	-	-	-
H		1	1/2		-	-	-	-
I		1	1/2		-	-	-	-

Table 5.7 Standard Interface Module Utilization by Mission (continued)

Sensor	LPS	RPS	HVS	PPS/UPS	DHU-S	DHU-C	MDSC-S	MDSC-C
SPACE TEL								
A		2	2		2	-		-
B	1	2	1	1	2	-	2	-
C		1	1		2	-		-
HEA-BLK 2								
A		1	1/2		-	-	-	-
B		1	1/2		-	-	-	-
C		1/2	1/2		-	-	-	-
D	1	1/2	1/2	1	-	-	-	-
E		1/2	1/2		-	-	-	-
F		1/2	1/2		-	-	-	-
G		1/2	1/2		-	-	-	-
H		1/2	1/2		-	-	-	-
ENV. MON								
A		1	1		-	-	-	-
B		1	1		-	-	-	-
C	1	1	1/2	1	-	-	-	-
D		1	1/2		2	-	2	-
E		1/2	1/2		2	-		-
F		1/2	1/2		-	-	-	-
EARTH SURVEY								
A	1	5	1	1	-	2	-	2
B		5	1		-	2	-	
SEOS								
A		2	1		2	-		-
B		2	1		2	-	2	-
C		1	-		2	-		-
D	1	1	-	1	2	-		-
E		1	-		-	-	-	-
F		1	-		-	-	-	-
G		1	-		-	-	-	-
H		1	-		-	-	-	-
SOLAR OBSER								
A		1	1/2		-	-	-	-
B		1	1/2		-	-	-	-
C		1/2	1/2		-	-	-	-
D	1	1/2	1/2	1	-	-	-	-
E		1/2	1/2		-	-	-	-
F		1/2	1/2		-	-	-	-
G		1/2	1/2		-	-	-	-
H		1/2	1/2		-	-	-	-

Table 5.8 Integral Design Recurring Hardware Utilization by Mission and Function Type

Sensor	Number of Missions	LPS	RPS	HVS	UPS/PPS	DHU-S	DHU-C	MDSC-S	MDSC-C
STORMSAT	4								
AASIR		4	4(E)*	-	4(B)	8	-	8	-
MASR		4	4(E)	-	4(B)	8	-	8	-
LANDSAT	8								
TMS		8	8(F)	-	8(B)	-	16	-	16
MSS		8	8(F)	8(C)	8(B)	-	16	-	16
TIROS	5								
AVHRR		5	-	-	-	-	-	-	-
TVOS									
BSU		5	-	-	5(B)	-	-	-	-
SSU		5	-	-	5(B)	-	-	-	-
MSU		5	-	-	5(B)	-	-	-	-
SEM									
TED		5	5(C)	5(D)	-	-	-	-	-
MEPED		5	5(C)	-	-	-	-	-	-
MEPED		5	5(C)	5(B)	-	-	-	-	-
UCS		5	5(C)	-	-	-	-	-	-
HEATE 1	1								
XRS		1	1(D)	1(E)	1(B)	-	-	-	-
XRC		1	1(C)	1(B)	1(B)	-	-	-	-
HEATE 2	1								
GSA		1	1(C)	1(B)	1(B)	-	-	-	-
CPA		1	1(C)	-	1(B)	-	-	-	-
GVE		1	1(C)	1(B)	1(B)	-	-	-	-
GPE		1	1(C)	1(C)	1(B)	-	-	-	-
GRE	1								
SCC		1	1(B)	1(C)	1(B)	-	-	-	-
TASC		1	1(C)	1(B)	1(B)	-	-	-	-
ACD		1	1(B)	1(B)	1(B)	-	-	-	-
TOFC		1	1(C)	1(B)	1(B)	-	-	-	-
SEASAT	2								
A		2	2(D)	2(B)	2(B)	-	-	-	-
B		2	2(C)	2(A)	2(A)	4	-	4	-
C		2	2(C)	2(B)	2(B)	-	-	-	-
D		2	2(C)	2(B)	2(A)	-	-	-	-
E		2	2(C)	2(B)	2(A)	-	-	-	-
F		2	2(C)	2(A)	2(A)	-	-	-	-
SOLAR MAX	2								
A		2	2(A)	2(A)	2(B)	-	-	-	-
B		2	2(A)	2(A)	2(B)	-	-	-	-
C		2	2(B)	-	2(B)	-	-	-	-
D		2	2(B)	2(A)	2(B)	-	-	-	-
E		2	2(B)	2(A)	2(B)	-	-	-	-
F		2	2(B)	2(A)	2(B)	-	-	-	-
G		2	2(B)	2(A)	2(B)	-	-	-	-
H		2	2(C)	2(A)	2(B)	-	-	-	-
I		2	2(C)	2(A)	2(B)	-	-	-	-

Table 5.8 Integral Design Recurring Hardware Utilization by Mission and Function Type (continued)

Sensor	Number of Missions	LPS	RPS	HVS	UPS/PPS	DHU-S	DHU-C	MDSC-S	MDSC-C
SPACE TEL	1								
A		1	1(D)	1(C)	1(B)	2	-	2	-
B		1	1(D)	1(B)	1(B)	2	-	2	-
C		1	1(D)	1(B)	1(B)	2	-	2	-
HEO BLK 2	3								
A		3	3(C)	3(A)	3(B)	-	-	-	-
B		3	3(C)	3(A)	3(B)	-	-	-	-
C		3	3(B)	3(A)	3(B)	-	-	-	-
D		3	3(B)	3(A)	3(B)	-	-	-	-
E		3	3(B)	3(A)	3(B)	-	-	-	-
F		3	3(B)	3(A)	3(B)	-	-	-	-
G		3	3(B)	3(A)	3(B)	-	-	-	-
H		3	3(B)	3(A)	3(B)	-	-	-	-
ENV. MON	2								
A		2	2(C)	2(B)	2(B)	-	-	-	-
B		2	2(C)	2(B)	2(B)	-	-	-	-
C		2	2(C)	2(B)	2(B)	-	-	-	-
D		2	2(C)	2(B)	2(B)	4	-	4	-
E		2	2(B)	2(A)	2(B)	-	-	-	-
F		2	2(B)	2(A)	2(B)	-	-	-	-
EARTH SURVEY	1								
A		1	1(F)	1(B)	1(B)	-	2	-	2
B		1	1(F)	1(B)	1(B)	-	2	-	2
SEOS	1								
A		1	1(D)	1(B)	1(B)	2	-	2	-
B		1	1(D)	1(B)	1(B)	2	-	2	-
C		1	1(C)	-	1(B)	2	-	2	-
D		1	1(C)	-	1(B)	2	-	2	-
E		1	1(C)	-	1(B)	-	-	-	-
F		1	1(C)	-	1(B)	-	-	-	-
G		1	1(C)	-	1(B)	-	-	-	-
H		1	1(C)	-	1(B)	-	-	-	-
SOLAR OBSER	1								
A		1	1(C)	1(A)	1(B)	-	-	-	-
B		1	1(C)	1(A)	1(B)	-	-	-	-
C		1	1(B)	1(A)	1(B)	-	-	-	-
D		1	1(B)	1(A)	1(B)	-	-	-	-
E		1	1(B)	1(A)	1(B)	-	-	-	-
F		1	1(B)	1(A)	1(B)	-	-	-	-
G		1	1(B)	1(A)	1(B)	-	-	-	-
H		1	1(B)	1(A)	1(B)	-	-	-	-

*The significance of (A)...(F) will be discussed in Section 5.2.11.

multipliers of the nonrecurring development for the integral design approach. To illustrate how Table 5.8 was derived, consider, for example, Stormsat.

Table 5.7 indicates that Stormsat will contain 2 sensors. With the standard interface module approach, the spacecraft will require one LPS servicing both sensors. The AASIR will require five RPS units, the MASR will require four RPS units, and both sensors will share a PPS/UPS unit. Two DHU-S units per sensor and two MDSC-S service both sensors in a redundant configuration. For an integral design approach, each sensor manufacturer will develop an LPS, RPS, PPS/UPS, DHU-S and MDSC-C, thereby incurring multiple nonrecurring costs for the same functions.

Table 5.8 lists the recurring quantities needed for an integral design approach. Again using Stormsat as an example, where one LPS standard interface module would be shared by two sensors, the equivalent of two LPS hardware functions would be provided by the two sensors. Four RPS modules are required for each Stormsat sensor to provide the total regulated power, one equivalent RPS would be required for each sensor with the physical characteristics adjusted accordingly, using the rationale of Paragraph 5.2.10 to provide the same capacity as four standard RPS modules. Similarly, one PPS/UPS equivalent would be required for each sensor, and a dual or redundant DHU/MDSC would be required per sensor for the integral design.

5.2.10 Physical Characteristic Adjustment

The standard interface modules have been configured for a specific level of performance capability. Where a sensor application requires greater than the prescribed performance level, multiple modules are required; conversely, where a single module has the capacity to serve multiple sensors, only a fraction of the module is assigned per sensor

(Table 5.7). To make use of the physical data derived for the modules to characterize the integral sensor design approach, the following assumptions were used:

- The multiple module assignments of Table 5.7 will scale the integral design hardware by 0.7. Using this relationship, if two RPS units are required to provide the regulated power for a sensor, the integral design hardware would only be 1.4 times the weight and size of the modular approach. The integral design efficiency will be realized by eliminating double packaging, connectors, mounting interfaces, etc.

- The fractional modular assignments of Table 5.7 will reduce the weight and size of the integral design hardware. The rationale is derived from the consideration that weight and volume of power supply components are linearly related to the dissipated power for the small range of fractional scaling considered (1/4-1).

- The integral HVS will have one-half the weight and volume of the standard module due to the simpler design for single voltage units.

- The DHU will interface directly with the MMS, no MDSC units are required. The DHU physical parameters are adjusted to contain remaining MDSC functions.

Using the above rationale, physical parameters for the hardware of the integral sensor design approach are shown in Table 5.9.

5.2.11 PRICE Program Results

The cost predictions for the hardware and quantities required for the standard interface module approach and the integral sensor design approach were obtained using the PRICE methodology and the hardware characterizations described in the previous sections. The results of the cost predictions are summarized in Tables 5.10 and 5.11. To determine the nonrecurring and recurring integral subsystem design costs for each mission, procurement Table 5.12

Table 5.9 Physical Characteristics of Integral Designs

Function	Configuration Definition					
	A	B	C	D	E	F
LPS						
Total Weight (kg)	.36	-	-	-	-	-
Total Volume (cm^3)	2832.0	-	-	-	-	-
Mechanical Weight (kg)	.05	-	-	-	-	-
RPS						
Total Weight (kg)	.36	.73	1.22	1.73	3.41	4.27
Total Volume (cm^3)	255.0	510.0	850.0	1189.0	2350.0	2945.0
Mechanical Weight (kg)	.04	.07	.12	.17	.34	.43
HVS						
Total Weight (kg)	.73	1.45	2.05	3.05	4.09	-
Total Volume (cm^3)	481.0	991.0	1416.0	2067.0	2775.0	-
Mechanical Weight (kg)	.09	.18	.27	.38	.50	-
UPS/PPS						
Total Weight (kg)	2.05	4.09	-	-	-	-
Total Volume (cm^3)	1416.0	2832.0	-	-	-	-
Mechanical Weight (kg)	.45	.91	-	-	-	-

Table 5.10 Standard Interface Module Characterization and Cost Summary

Module	Wt., (kg)	Vol., (cm^3)	Qty	Nonrecurring Cost (K$)			Recurring Cost (K$)				Total Cost (K$)
				Eng	Prod	Total	Eng	Prod	Total Rec	Unit Cost	
RPS	1.23	850	311	12	48	59	169	2666	2835	9.12	2894
LPS	1.23	850	79	12	48	59	77	805	882	11.17	942
HVS	1.45	991	141	32	47	80	111	1461	1572	11.15	1652
PPS/UPS	4.09	2832	85	29	109	138	167	1955	2122	24.96	2260
DHU-S	.80	991	73	113	130	243	216	1612	1829	25.05	2072
DHU-C	.80	991	45	265	100	365	205	1141	1346	29.91	1711
MDSC-S	.60	6512	41	98	119	217	175	898	1073	26.17	1290
MDSC-C	.60	6512	24	222	84	306	160	561	721	30.04	1027

Table 5.11 Integral Design Cost Summary (Thousands of dollars)

Mission	LPS		RPS		HVS		PPS/UPS		DHU-S		DHU-C	
	Nonrec	Rec	Nonrec	Rec	Nonrec	Rec	Nonrec	Rec	Nonrec	Rec	Nonrec	Rec
STORMSAT	20	48	184	396	-	-	138	356	340	1094	-	-
LANDSAT	20	86	224	958	52	169	138	640	-	-	504	1932
TIROS	80	232	152	392	114	253	345	639	-	-	-	-
HEAT-1	20	26	102	106	135	118	138	164	-	-	-	-
HEAT-2	40	52	152	160	132	118	276	328	-	-	-	-
GRE	40	52	124	132	172	154	276	328	-	-	-	-
SEASAT	60	90	241	322	206	228	290	432	170	320	-	-
SOLAR MAX	90	135	222	294	184	208	621	918	-	-	-	-
SPACE TEL	30	39	153	159	132	118	207	246	510	585	-	-
HEO BLK 2	80	152	220	392	184	272	552	1112	-	-	-	-
ENV. MON	60	90	200	268	206	222	414	612	170	320	-	-
EARTH SURVEY	20	26	224	214	80	72	138	164	-	-	504	390
SEOS	80	104	330	346	80	72	552	656	680	780	-	-
SOLAR OBSERV	80	104	220	236	184	176	552	656	-	-	-	-

Table 5.12 Integral Design Procurement Table									
Function	Quantity								
	1	2	3	4	5	6	7	8	16
LPS	X	X	X	X	X			X	
RPS									
A (1/4)			X					X	
B (1/2)	X	X	X	X					
C (1)	X	X	X		X				
D (2)	X	X							
E (4)				X					
F (5)	X							X	
HVS									
A (1/2)	X	X	X						
B (1)	X	X			X				
C (2)	X							X	
D (3)					X				
E (4)	X								
UPS/PPS									
A (1/2)	X	X							
B (1)	X	X	X	X	X			X	
DHU-S		X		X				X	
MDSC-S		X		X				X	
DHU-C		X							X
MDSC-C		X							X

was derived from the data given in Table 5.8. Table 5.8 shows for Stormsat, for example, that each of the two sensors will have associated nonrecurring costs to develop the LPS function, an RPS function equivalent to four RPS SIM modules ('E' configuration of Table 5.9) and a UPS/PPS function equivalent to a UPS/PPS SIM module ('B' configuration of Table 5.9) and a redundant DHU-S and MDSC-S. Each sensor will have recurring costs for these hardware elements based upon a buy of four, one for each mission. A single prototype was assumed and spares would be provided at the piece part or board level. Each PRICE program output consists of a print-out of the input data, along with the cost prediction for various program elements. A complete set of PRICE results and input data sheets are contained in Section 8.2 and 8.3. Section 8.2 presents the SIM related material and Section 8.3 presents the sensor subsystem (integral functions) related material. The description of the PRICE input data terminology is given in Section 8.1.

5.3 Benefit Determination for MMS Missions

The economic analysis is concerned with using the unit recurring and nonrecurring costs developed in Section 5.2 to establish the annual cost of performing the mission model with and without the utilization of the standard interface modules as defined in Section 4.0. The economic analysis is also concerned with comparing these annual costs and developing the economic benefits, i.e., cost savings, which may result from the utilization of the standard interface modules. Finally, the economic analysis is concerned with an assessment of the desirability of developing the standard interface modules in terms of the benefits, the investment and the risk.

The basic inputs to the economic analysis are the mission model (as delineated in Figure 3.1), the standard interface module utilization by mission (as detailed in Table 5.7), the integral design recurring hardware utilization by mission and function type (as described in Table 5.9), the standard interface module unit recurring and nonrecurring cost (as given in Table 5.10), the integral design unit recurring and nonrecurring cost by mission and function type (as given in Table 5.11), and the number of standard interface module spares, including nonflight ground equipment (as described in Table 4.13). These data are combined in Tables 5.13 through 5.18 to yield the annual recurring cost of the research and development and the applications missions, the nonrecurring cost and the annual cost of spares for each of the considered standard interface modules (the DHU-S and MDSC-S are considered together as are the DHU-C and MDSC-C since these are always used in combination on the applicable missions). When using the various SIM, it should be noted that the nonrecurring cost is mission independent. This is to be contrasted with the integral design approach (as can be seen by referring to Tables 5.19 through 5.24) where the nonrecurring costs are a direct function of the number of missions. In other words, the SIM nonrecurring cost is incurred once for the totality of the missions considered whereas the integral design hardware nonrecurring cost is incurred once for each mission.

The determination of the annual cost of the various SIM alternatives assumes that all costs are incurred instantaneously (i.e., no cost spreading) with all recurring costs occurring in the year of the applicable flight. The annual cost of spares is based upon acquiring all necessary spares (including ground support equipment and flight and ground spares) over a three-year period.

The determination of the annual costs associated with the integral design hardware (alternative to the SIM approach) is given in Tables 5.19 to 5.24. It should be noted that nonrecurring costs are incurred once for each mission. It is assumed that the nonrecurring costs are

Table 5.13 Standard Interface Module Annual Cost (LPS Function) (in 1976 K$)

Mission	Fiscal Year				
	1981	1982	1983	1984	1985
HEATE(II)	11.2				
Solar Max.			11.2		11.2
Space Tele.			11.2		
HEAO BLK II				11.2	22.4
Solar Observ.					11.2
StormSat	11.2			11.2	
TIROS	11.2		11.2		
SeaSat		11.2			11.2
Environ. Mon.		11.2		11.2	
Landsat	11.2	11.2		11.2	11.2
SEOS					11.2
Earth Survey			11.2		
Rec. Cost	44.8	33.6	44.8	44.8	78.4
NOAA		11.2	11.2	11.2	
GOES			11.2	11.2	
Earth Resources		11.2	11.2	11.2	11.2
Rec. Cost		22.4	33.6	33.6	11.2
Nonrec. Cost	59.0	-	-	-	-
Cost of Spares	179.2	179.2	179.2	-	-
Total Cost	283.0	235.2	257.6	78.4	89.6

Table 5.14 Standard Interface Module Annual Cost (RPS Function) (in 1976 K$)

Mission	Fiscal Year				
	1981	1982	1983	1984	1985
HEATE(II)	27.4				
Solar Max.			45.6		45.6
Space Tele.			45.6		
HEAO BLK II				45.6	91.2
Solar Observ.					45.6
StormSat	73.0			73.0	
TIROS	45.6		45.6		
SeaSat		63.8			63 8
Environ. Mon.		45.6		45.6	
Landsat	91.2	91.2		91.2	91.2
SEOS					91.2
Earth Survey				91.2	
Rec. Cost	237.2	200.6	228.0	255.4	428.6
NOAA		45.6	45.6	45.6	
GOES			73.0	73.0	
Earth Resources		91.2	91.2	91.2	91.2
Rec. Cost	-	136.8	209.8	209.8	91.2
Nonrec. Cost	59.0	-	-	-	-
Cost of Spares	273.6	264.5	264.5	-	-
Total Cost	569.8	601.9	702.3	465.2	519.8

Table 5.15 Standard Interface Module Annual Cost (HVS Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE(II)	44.6				
Solar Max.			44.6		44.6
Space Tele.			44.6		
HEAO BLK II				44.6	89.2
Solar Observ.					44.6
StormSat	-			-	
TIROS	44.6		44.6		
SeaSat		55.8			55.8
Environ. Mon.		55.8		55.8	
Landsat	22.3	22.3		22.3	22.3
SEOS					22.3
Earth Survey			22.3		
Rec. Cost	111.5	133.9	156.1	122 7	278.8
NOAA		44.6	44.6	44.6	
GOES			-	-	
Earth Resources		22.3	22.3	22.3	22.3
Rec. Cost		66.9	66.9	66.9	22.3
Nonrec. Cost	80.0	-	-	-	-
Cost of Spares	189.6	189.6	178.4		
Total Cost	381.1	390.4	401.4	189.6	301.1

Table 5.16 Standard Interface Module Annual Cost (PPS/UPS Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE(II)	25.0				
Solar Max.			25.0		25.0
Space Tele.			25.0		
HEAO BLK II				25.0	50.0
Solar Observ.					25.0
StormSat	25.0			25.0	
TIROS	75.0		75.0		
SeaSat		100.0			100.0
Environ. Mon.		25.0		25.0	
Landsat	25.0	25.0		25.0	25.0
SEOS					25.0
Earth Survey			25.0		
Rec. Cost	150.0	150.0	150.0	100.0	250.0
NOAA		75.0	75.0	75.0	
GOES			25.0	25.0	
Earth Resources		25.0	25.0	25.0	25.0
Rec. Cost		100.0	125.0	125.0	25.0
Nonrec. Cost	138.0	-	-	-	-
Cost of Spares	400.0	400.0	400.0	-	-
Total Cost	688.0	650.0	675.0	225.0	275.0

Table 5.17 Standard Interface Module Annual Cost (DHU-S plus MDSC-S Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE(II)	-				
Solar Max.			-		-
Space Tele.			202.6		
HEAO BLK II				-	-
Solar Observ.					
StormSat	152.5			152.5	
TIROS	-		-		
SeaSat		102.4			102.4
Environ. Mon.		102.4		102.4	
LandSat	-	-		-	-
SEOS					252.7
Earth Survey			-		
Rec. Cost	152.5	204.8	202.6	254.9	355.1
NOAA		-	-	-	
GOES			152.5	152.5	
Earth Resources		-	-	-	
Rec. Cost	-	-	152.5	152.5	-
Nonrec. Cost	460.0	-	-	-	-
Cost of Spares	253.9	253.9	253.9	-	-
Total Cost	866.4	458.7	609.0	407.4	355.1

Table 5.18 Standard Interface Module Annual Cost (DHU-C plus MDSC-C Function) (in 1976 K$)

Mission	Fiscal Year				
	1981	1982	1983	1984	1985
HEATE(II)	-				
Solar Max.			-		-
Space Tele.			-		
HEAO BLK II				-	-
Solar Observ.					-
StormSat	-			-	
TIROS	-		-		
SeaSat		-			-
Environ. Mon.		-		-	
Landsat	179.7	179.7		179.7	179.7
SEOS					-
Earth Survey			179.7		
Rec. Cost	179.7	179.7	179.7	179.7	179.7
NOAA		-	-	-	
GOES			-	-	
Earth Resources		179.7	179.7	179.7	179.7
Rec. Cost	-	179.7	179.7	179.7	179.7
Nonrec. Cost	671.0	-	-	-	-
Cost of Spares	179.7	179.7	149.7	-	-
Total Cost	1030.4	539.1	509.1	359.4	359.4

Table 5.19 Integral Design Hardware Annual Cost (LPS Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE(II)	40/52				
Solar Max.			90/68		-/67
Space Tele.			30/39		
HEAO BLK II				80/51	-/101
Solar Observ.					80/104
StormSat	20/12			-/12	
TIROS	80/46		-/46		
SeaSat		60/45			-/45
Environ. Mon.		60/45		-/45	
LandSat	20/11	-/11		-/11	-/11
SEOS					80/104
Earth Survey			20/26		
Sub Total*	281	221	319	199	592
NOAA		-/46	-/46	-/46	
GOES			-/12	-/12	
Earth Resources		-/11	-/11	-/11	-/11
Sub Total*		57	69	69	11
Total*	281	278	388	268	603

Note: xx/yy indicates annual nonrecurring (xx) and recurring (yy) costs

*Sum of nonrecurring and recurring costs.

Table 5.20 Integral Design Hardware Annual Costs (RPS Function) (in 1976 K$)

Mission	Fiscal Year				
	1981	1982	1983	1984	1985
HEATE(II)	152/160				
Solar Max.			222/147		-/147
Space Tele.			153/159		
HEAO BLK II				220/131	-/262
Solar Observ.					220/232
StormSat	184/99			-/99	
TIRCS	152/78		-/78		
SeaSat		241/161			-/161
Environ. Mon.		200/134		-/134	
LandSat	224/120	-/120		-/120	-/120
SEOS					330/346
Earth Survey			224/214		
Sub Total*	1169	856	1197	704	1818
NOAA		-/78	-/78	-/78	
GOES			-/99	-/99	
Earth Resources		-/120	-/120	-/120	-/120
Sub Total*		198	297	297	120
Total*	1169	1054	1494	1001	1938

Note: xx/yy indicates annual nonrecurring (xx) and recurring (yy) costs

*Sum of nonrecurring and recurring costs

Table 5.21 Integral Design Hardware Annual Costs (HVS Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE (II)	132/118				
Solar Max.			184/104		-/104
Space Tele.			132/118		
HEAO BLK II				184/91	-/182
Solar Observ.					184/176
StormSat	-/-			-/-	
TIROS	114/51		-/51		
SeaSat		206/114			-/114
Environ. Mon.		206/111		-/111	
Landsat	52/21	-/21		-/21	-/21
SEOS					80/72
Earth Survey			80/72		
Sub Total*	488	658	741	407	933
NOAA		-/51	-/51	-/51	
GOES			-/-	-/-	
Earth Resources		-/21	-/21	-/21	-/21
Sub Total*		72	72	72	21
Total*	488	730	813	479	933

Note: xx/yy indicates annual nonrecurring (xx) and recurring (yy) costs

*Sum of nonrecurring and recurring costs

Table 5.22 Integral Design Hardware Annual Costs (PPS/UPS Function) (in 1976 K$)

Mission	Fiscal Year				
	1981	1982	1983	1984	1985
HEATE (II)	276/328				
Solar Max.			621/459		-/459
Space Tele.			207/246		
HEAO BLK II				552/370	-/741
Solar Observ.					552/656
StormSat	138/89			-/89	
TIROS	345/128		-/128		
SeaSat		290/216			-/216
Environ. Mon.		414/306		-/306	
LandSat	138/80	-/80		-/80	-/80
SEOS					552/656
Earth Survey			138/164		
Sub Total*	1522	1306	1963	1397	3912
NOAA		-/128	-/128	-/128	
GOES			-/89	-/89	
Earth Resources		-/80	-/80	-/80	-/80
Sub Total*		208	297	297	80
Total*	1522	1514	2260	1694	3992

Note: xx/yy indicates annual nonrecurring (xx) and recurring (yy) costs

*Sum of nonrecurring and recurring costs

Table 5.23 Integral Design Hardware Annual Costs (DHU-S Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE II	-/-				
Solar Max.			-/-		-/-
Space Tele.			510/585		
HEAO BLK II				-/-	-/-
Solar Observ.					-/-
StormSat	340/274			-/274	
TIROS	-/-		-/-		
SeaSat		170/160			-/160
LandSat	-/	-/-		-/-	-/-
SEOS					680/780
Earth Survey			-/-		
Sub Total*	614	660	1095	434	1620
NOAA		-/-	-/-	-/-	
GOES			-/274	-/274	
Earth Resources		-/-	-/-	-/-	-/-
Sub Total*			274	274	
Total*	614	660	1369	708	1620

Note: xx/yy indicates annual nonrecurring (xx) and recurring (yy) costs

*Sum of nonrecurring and recurring costs

Table 5.24 Integral Design Hardware Annual Costs (DHC-S Function) (in 1976 K$)

Mission	Fiscal Year 1981	1982	1983	1984	1985
HEATE (II)	-/-				
Solar Max.			-/-		-/-
Space Tele.			-/-		
HEAO BLK II				-/-	-/-
Solar Observ.					-/-
StormSat	-/-			-/-	
TIROS	-/-		-/-		
SeaSat		-/-			-/-
Environ. Mon.		-/-		-/-	
LandSat	504/242	-/242		-/242	-/242
SEOS					-/-
Earth Survey			504/390		
Sub Total*	746	242	894	242	242
NOAA		-/-	-/-	-/-	
GOES			-/-	-/-	
Earth Resources		-/242	-/242	-/242	-/242
Sub Total*		242	242	242	242
Total*	746	484	1136	484	484

Note: xx/yy indicates annual nonrecurring (xx) and recurring (yy) costs.

associated with the first flight of each mission with no additional nonrecurring costs being incurred on succeeding flights.

Tables 5.25 and 5.26 summarize the annual costs of the integral hardware design and SIM alternatives and develop the annual savings of the SIM alternative relative to the integral hardware design alternative. Figure 5.5 summarizes the annual cost of the integral design and SIM alternatives. Table 5.25 is based upon the nominal number of spares as developed in Table 4.13 whereas Table 5.26 is based upon 1.5 times the nominal number of spares. The impact of increasing the number of spares is twofold, namely a decrease in unit recurring cost because of increased quantities (i.e., learning effects) and an increase in annual costs because of the increased number of spares. In general the impact of changing the level of spares by $\pm$ 50 percent results in a unit cost change of -2 percent to +6 percent.

The net present value (NPV) of the savings cash flow streams (i.e., the present value of the annual cost of the integral hardware design alternative less the present value of the annual cost of the SIM alternative) are summarized at two different levels of spares in Table 5.27. The reason for considering the different levels of spares, is that the sparing philosophy has not been worked out in detail and the cost of spares is a very definite penalty against the SIM alternatives. The impact of the cost of spares can best be seen from the benefit/cost ratio (B/C). The benefit/cost ratio is defined as the net present value divided by the present value of the cost (i.e., the investment required to achieve the benefits). PVC is therefore the present value of the negative cash flow stream.

Table 5.28 illustrates the impact of the cost of ground support units on the net savings due to the SIM alternative. It should be noted that the savings range from $17.6

Table 5.25 Annual Cost and Savings Summary (Nominal Spares)
(thousands of 1976 dollars)

Function		Fiscal Year 1981	1982	1983	1984	1985	Total
LPS	Integral	281.0	278.0	388.0	268.0	603.0	1818
	SIM	283.0	235.2	257.6	78.4	89.6	944
	Savings	-2.0	42.8	130.4	189.6	513.4	874
RPS	Integral	1169.0	1054.0	1494.0	1001.0	1938.0	6656
	SIM	569.8	601.9	702.3	465.2	519.8	2859
	Savings	599.2	452.1	791.7	535.8	1418.2	3797
HVS	Integral	488.0	730.0	813.0	479.0	933.0	3443
	SIM	381.1	390.4	401.4	189.6	301.1	1664
	Savings	106.9	339.6	411.6	289.4	631.9	1779
PPS/UPS	Integral	1522.0	1514.0	2260.0	1694.0	3992.0	10982
	SIM	688.0	650.0	675.0	225.0	275.0	2513
	Savings	834.0	864.0	1585.0	1469.0	3717.0	8469
DHU-S + MDSC-S	Integral	614.0	660.0	1369.0	708.0	1620.0	4971
	SIM	866.4	458.7	609.0	407.4	355.1	2697
	Savings	-252.4	201.3	760.0	300.6	1264.9	2274
DHU-C + MDSC-C	Integral	746.0	484.0	1136.0	484.0	484.0	3334
	SIM	1030.4	539.1	509.1	359.4	359.4	2797
	Savings	-284.4	-55.1	626.9	124.6	124.6	537

Table 5.26 Annual Cost and Savings Summary (1.5^{x} Nominal Spares)
(Thousands of 1976 Dollars)

Function		Fiscal Year 1981	1982	1983	1984	1985
LPS	Integral	281.0	278.0	388.0	268.0	603.0
	SIM	372.0	324.8	347.2	78.4	89.6
	Savings	-91.0	-46.8	40.8	189.6	513.4
RPS	Integral	1169.0	1054.0	1494.0	1001.0	1938.0
	SIM	706.6	734.2	834.6	465.2	519.8
	Savings	462.4	319.8	659.4	535.8	1418.2
HVS	Integral	488.0	730.0	813.0	479.0	933.0
	SIM	477.9	485.2	490.6	189.6	301.1
	Savings	10.1	244.8	322.4	289.4	631.9
PPS/UPS	Integral	1522.0	1514.0	2260.0	1694.0	3992.0
	SIM	888.0	850.0	875.0	225.0	275.4
	Savings	634.0	664.0	1385.0	1469.0	3717.0
DHU-S	Integral	614.0	660.0	1369.0	708.0	1620.0
+	SIM	993.4	585.7	736.0	407.4	355.1
MDSC-S	Savings	-379.4	74.3	633.0	300.6	1264.9
DHU-C	Integral	746.0	484.0	1136.0	484.0	484.0
+	SIM	1120.3	629.0	584.0	359.4	359.4
MDSC-C	Savings	-374.3	-145.0	552.0	124.6	124.6

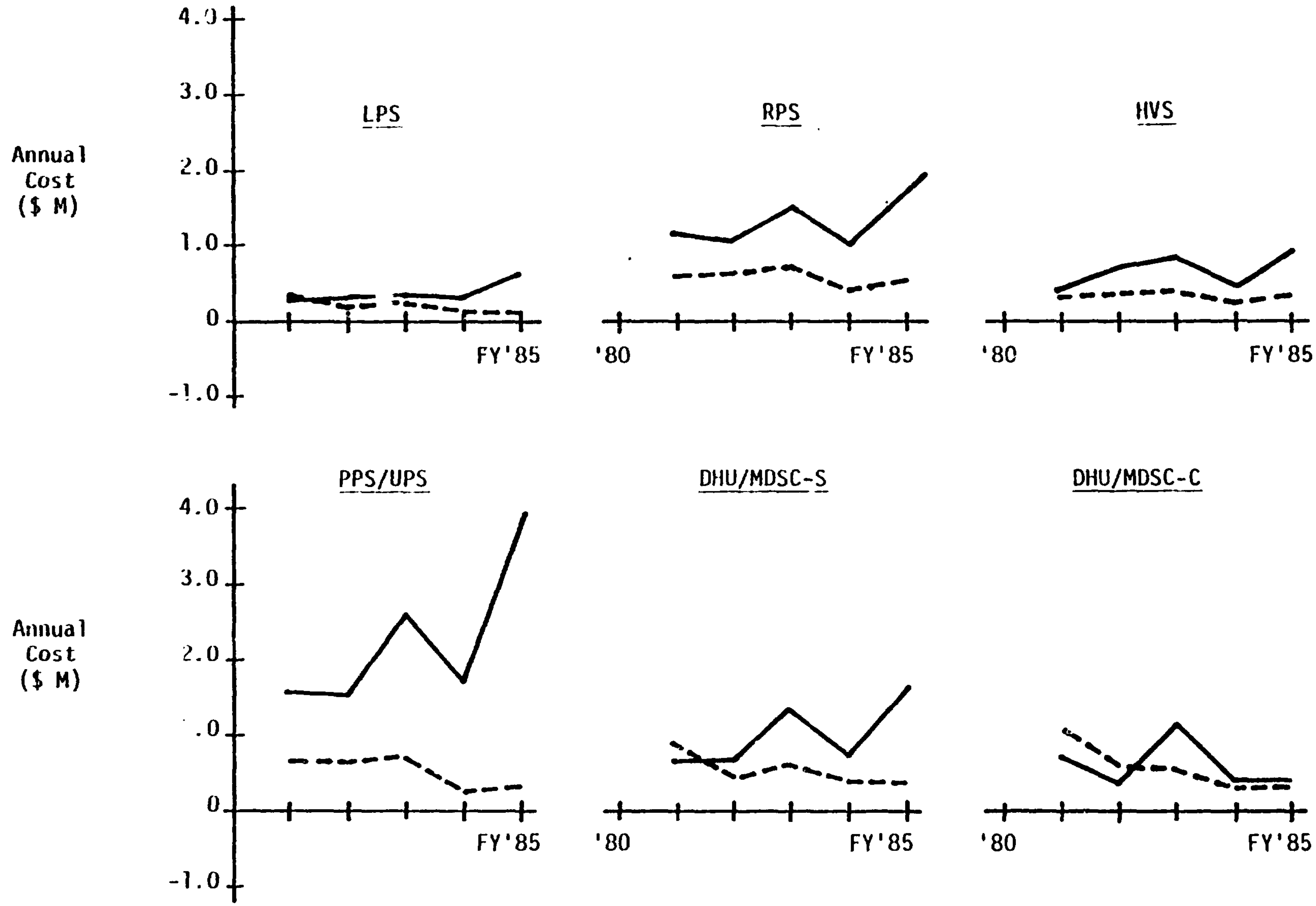


Figure 5.5 Summary of the Annual Costs of the Integral Design (—) and SIM (----) Alternatives (Nominal Spares)

Table 5.27 Net Present Value and Benefit/Cost Ratio of SIM

Function	Nominal Spares			1.5^{x} Nominal Spares		
	NPV(K$)	PVC(K$)	B/C	NPV(K$)	PVC(K$)	B/C
LPS	436	1.4	320	269	91	3.0
RPS	2071	0	∞	1823	0	∞
HVS	953	0	∞	780	0	∞
PPS/UPS	4489	0	∞	4115	0	∞
DHU-S + MDSC-S	1126	172	6.6	889	258	3.5
DHU-C + MDSC-C	247	228	1.1	88	344	.3

Table 5.28 Net Savings (K $) of SIM Alternative Relative to Integral Design Alternative Over 1981-1985 Time Period

Function	Ground Support Unit Cost as a Function of Flight Unit Recurring Cost					
	1.0	.8	.6	.4	.2	0
LPS	874	972	1071	1169	1267	1365
RPS	3797	3916	4034	4153	4271	4390
HVS	1698	1787	1876	1966	2055	2144
PPS/UPS	8469	8689	8908	9128	9347	9567
DHU/MDSC-S	2274	2375	2477	2578	2679	2781
DHU/MDSC-C	537	609	681	753	824	896
Total	17649	18348	19047	19747	20443	21143

million to $21.1 million as a function of the cost of the ground support units. Figure 5.6 illustrates the effect of the cost of ground support units on the net present value associated with each SIM. Reduction of the cost of ground support units for the SIM increases the NPV and makes each of the SIMs a more attractive investment opportunity.

Indications are that each of the SIM alternatives is a worthwhile undertaking since the net present value (at 10 percent discount rate) of each is positive. However, the desirability of developing the SIM must also include considerations of uncertainty and risk. The range of uncertainties such as costs, procurement philosophy, sensor requirements, etc., have not been considered in this analysis. An attempt was made to indicate the sensitivity of the results (i.e., benefits) to level of spares with the conclusion that the RPS, HVS, and PPS/UPS standard interface module economic benefits are not appreciably affected even by relatively large variations in the level of spares required. In particular, the net present values of the RPS and PPS/UPS are so large that the development of these SIM should be considered as low risk programs.

A major area of uncertainty is the mission model--that is, the missions which will ultimately utilize the standard interface modules given that they are developed. Tables 5.29 and 5.30 summarize the SIM economics in terms of the number of missions which might utilize the developed SIM. First it must be pointed out that if the number of missions for which each SIM is applicable increases above that indicated in Table 5.28, then it is clear that all of the indicated SIMs should be developed (with the possible exception of the DHU-C and MDSC-C). This is true because of the following reasons: (a) the SIM nonrecurring cost is independent of the number of missions whereas the integral design hardware nonrecurring costs increase with the number of missions,

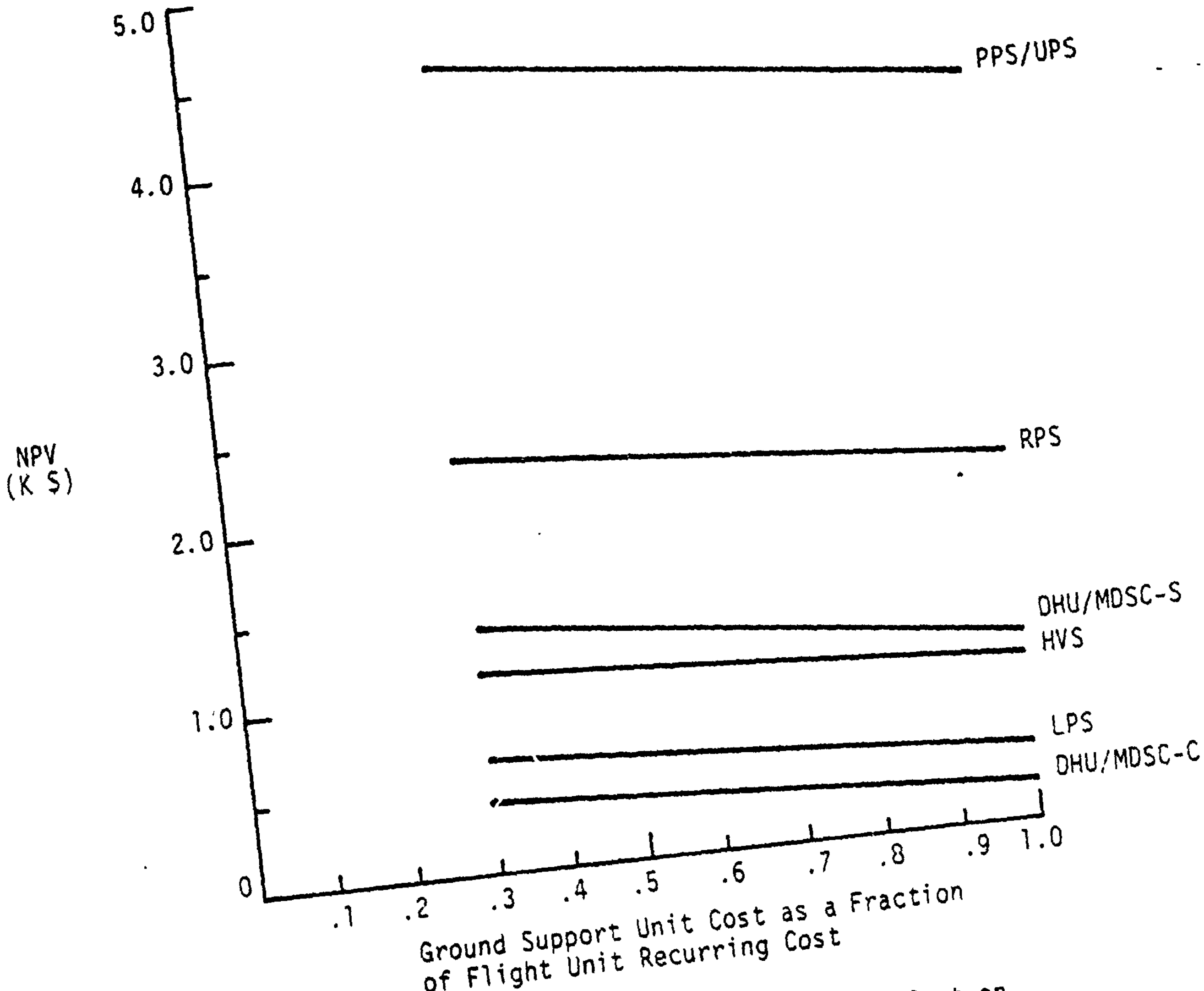


Figure 5.6 Impact of Ground Support Unit Cost on Net Present Value of Savings (Nominal Spares)

Table 5.29 Summary of SIM Economics (Nominal SIM Mission Model)

Function	No. of Missions Using SIM*	No. of Flights Using SIM**	SIM NRC (K$)	SIM RC+ (K$)	NRC Foregone (K$)	RC Foregone (K$)	ANRC** (K$)	ARC** (K$)
LPS	12	31	59	885	660	1158	+601	+273
RPS	12	31	59	2800	2522	4134	+2463	+1334
HVS	11	27	80	1584	1554	1889	+1474	+305
PPS/UPS	12	31	138	2375	4223	6759	+4085	+4384
DHU-S + MDSC-S	5	12	460	2237	1870	3101	+1410	+864
DHU-C + MDSC-C	2	13	671	2126	1008	2326	+337	+200

* Not including Applications Missions
** Including Applications Missions
\+ Including Cost of spares (nonflight equipment)
\++ Reduction in Cost (+ indicates cost reduction due to SIM)

Table 5.30 Summary of SIM Economics (Reduced SIM Mission Model)

Function	No. of Missions Using SIM*	No. of Flights Using SIM**	SIM NRC (K$)	SIM RC+ (K$)	NRC Foregone (K$)	RC Foregone (K$)	ΔNRC** (K$)	ΔRC** (K$)
LPS	6	16	59	456	340	598	+281	+142
RPS	6	16	59	1445	1302	2134	+1243	+689
HVS	6	14	80	821	806	979	+726	+158
PPS/UPS	6	16	138	1226	2180	3488	+2042	+2262
DHU-S + MDSC-S	3	6	460	1118	935	1550	+475	+432
DHU-C + MDSC-C	1	7	671	1144	543	1252	-128	+108

* Not Including Applications Missions
** Including Applications Missions
\+ Including Cost of Spares (nonflight equipment)
\+\+ Reduction in Cost (+ indicates cost reduction due to SIM)

and (b) since SIM unit recurring costs are less than the integral design unit recurring costs, then additional recurring cost savings will be achieved as the number of missions increases.

The impact of reducing the number of missions can be seen from Tables 5.29 and 5.30. In these tables, NRC is nonrecurring cost, RC is recurring cost, NRC and RC "foregone" are the nonrecurring and recurring costs, respectively, of the integral design hardware alternatives that can be eliminated if the SIM alternatives are developed, and ΔNRC and ΔRC are the nonrecurring and recurring cost savings that can be achieved if the SIMs are developed. Note that in general, most of the savings are in the area of reduced nonrecurring costs. This indicates that the results may be sensitive to the number of missions which will use the SIM. This is indeed the case where one-half the nominal mission model is considered, as is indicated in Table 5.30. Note that when the number of missions is reduced by a factor of two, the DHU-C plus MDSC-C SIM is not economically desirable. In the case of the DHU-C plus MDSC-C, only two missions in the nominal mission set have need for this SIM. Thus a factor of two reduction implies that one of the two missions will not use the DHU-C plus MDSC-C. Since this is a distinct possibility, there is a reasonably high level of risk associated with the development of the DHU-C and MDSC-C. Indeed, the state of the art in digital data handling equipment is moving at such a pace that the "complex" DHU and MDSC may end up as a minor variation of the "simple" DHU and MDSC by the time the sensors are ready to fly.

Thus, it is tentatively concluded that all of the SIM considered herein should be developed with the possible exception of the DHU-C and MDSC-C. It must be emphasized that the analyses reported herein did not attempt to optimize the design of the various SIM from the points of view of level of modularity, procurement philosophy, level of spares and ground equipment, etc. It is felt that optimization will tend, in general, to increase the magnitude of estimated economic benefits and will, therefore reinforce the above conclusions.

5.4 Extension of the Benefit Determinations to Other Missions

The specific emphasis of this study is the analysis of the economic benefits attributed to the use of the SIM in conjunction with the MMS. Thus, in this study, only the specific SIM requirements of the MMS have been developed and analyzed. However, it is possible to obtain an ad hoc estimate of the economic benefits that could accrue through the use of the SIM in conjunction with other spacecraft by extrapolating the result obtained in the analysis of the use of the SIM in MMS missions. It must be cautioned that the accuracy of these extrapolations is such that they indicate only rough order of magnitude benefits which require further corroboration through more detailed analysis.

The composite payload planning model (Figure 3.1) used as the basis for this study indicates that a total of 125 separate and distinct flights are planned for the period 1981 through 1985. These include 31 flights of the MMS, 54 Spacelab flights, and 40 flights of the SMMS and other spacecraft. A preliminary examination of these 40 flights indicates that 12 may be performed by an SMMS. The balance of 28 flights consists of geostationary, planetary, and miscellaneous missions that are not now considered to be candidates for use of the MMS, SMMS, or Spacelab.

In the absence of a detailed study of the possibilities for the use of SIM in these non-MMS missions, it is necessary to make certain assumptions in order to extrapolate the benefit estimate to the balance of the payload planning model. Firstly, it is necessary to assume that the SIM used in the non-MMS missions are identical to those designed for use in the MMS missions. This assumption probably introduces a degree of conservatism into the resulting benefit estimates, as the number of types of functions that become candidates for standardization will probably increase with the increasing number of missions, thus making additional types of SIMs both technically feasible and economically attractive. Secondly, it is necessary to make assumptions concerning the quantity of SIM used in the non-MMS missions. Thirdly, the possible reuse of flight hardware in the Spacelab missions must be considered. While reasonable "order of magnitude" assumptions can be made in the latter two cases, it is important to note that none of these three assumptions have received detailed attention in this study.

Table 5.31 lists the use (U) and reuse (R) factors assumed for the estimation of benefits associated with other missions. The use factor (U) is defined as the ratio of the number of SIM used in a non-MMS mission to the average number used in an MMS mission. The reuse factor (R) is defined as the average number of times that a flight SIM is used, as a result of recovery and reuse. It should be noted that the concept of reuse is only applicable to the Spacelab missions in the 1981-1985 time frame considered for this study, and that it is assumed that both standard (SIM) and nonstandard flight hardware will be capable of reuse in subsequent missions. The incremental costs of refurbishment and retest have not been included.

Using the data shown in Table 5.28, for a total of 31 MMS flights, the average cost saving per MMS flight that is obtained through the use of the SIM ($\bar{C}$) is in the range of \$569,322 to \$682,032. The total benefit across the payload planning model is then determined by the relationship:

$$B = \bar{C} \sum_i \frac{N_i U_i}{R_i}$$

Using the above relationship, the total cost savings across the entire composite payload planning model for the period 1981 to 1985 is estimated to be in the range of \$65.6 million to \$78.4 million. The contribution of each class of missions to this aggregate savings is shown in Table 5.32.

Considering the important contribution of the Spacelab mission to those estimated savings, it is of interest to note the effect of a changed set of Spacelab assumptions on the results. If the assumed usage of SIM in the Spacelab missions is doubled, and the reuse factor decreased from R=4 to R=3, the savings for the Spacelab missions are increased to $88.5 million to $106.1 million, and the aggregate savings across the composite payload planning model becomes $113.9 million to $143.5 million.

Table 5.31 SIM Use and Reuse Factors for Non-MMS Missions

Missions	Number of Flights*, N	Use Factor, U	Reuse Factor, R
MMS	31	1.0	1
SMMS	12	0.25	1
Other	28	0.75	1
Spacelab			
OSS	5	10.	4
Vesibular Function Research	1	10.	4
Life Science Carry-ons	11	0.25	4
Earth Viewing Applications Lab	6	10.	4
Space Processing Lab	7	10.	4
OAST	24	2.	4
TOTAL	125	-	-

*1981-1985

Table 5.32 Savings from Use of SIM for Each Spacecraft Class Considered

Spacecraft	No. of Flights	Range of Savings ($ Millions)
MMS	31	17.7 to 21.1
SMMS	12	1.7 to 2.0
Other	28	12.0 to 14.3
Spacelab	54	34.2 to 41.0
Total	125	65.6 to 78.4

6. CONCLUSIONS

This study of the potential application of SIM for use with the MMS has identified the technical and economic feasibility of standardizing power conditioning and data handling functions. A requirement was determined for four types of standardized power modules, as follows:

- High Voltage Supply (HVS)
- Logic Power Supply (LPS)
- Regulated Analog Power Supply (RPS)
- Pulsed and Unregulated Power Supply (PPS/UPS)

Two levels of complexity were identified for a standardized, modular data processing system consisting of sensor data handling units (DHU), a master data sequencer and controller (MDSC), and a standard control and data bus. For the purpose of this study, units with single channel data handling rates up to one Mbps were designated as simple (DHU-S, MDSC-S), while units requiring greater than this bit rate were designated as complex (DHU-C, MDSC-C).

The use of these SIM in the 31 MMS flights anticipated in the 1981 to 1985 time period could result in a net cost savings to NASA in the range of $17.7 million to $21.1 million. When compared to nonstandard interface hardware, and including nonrecurring development costs, each SIM type is found to produce a cost savings, and each type considered has a benefit-to-cost ratio greater than one. However, consideration of possible spares philosophies and possible changes in the mission model appear to make the complex version of the modular data processing system (DHU-C, MDSC-C) marginal from an economic standpoint. Thus, it is tentatively concluded that all of the SIM considered in this study, with the possible exception of the DHU-C and MDSC-C, should be developed.

Consideration of the extension of the use of the SIM concept identified in this study for possible use with the MMS to the SMMS, Spacelab, and other spacecraft missions contemplated for the 1981-1985 time period leads to an estimate of total net cost savings in the range of $65 million to $143.5 million. This range of cost savings results from several important factors that cannot be determined with certainty at the present time. Reasons for this range in the value of cost savings include uncertainties in the spares philosophy, the cost of ground support SIM, the extent of possible use of SIM in Spacelab missions, as well as the number of possible reuses of flight hardware SIM in Spacelab missions. It is noted that this estimate of cost savings across all of the missions could be somewhat understated, as the possibilities for standardization should increase with increasing numbers of missions.

7. RECOMMENDATIONS FOR FUTURE WORK

The results of this study indicate that the development of power conditioning and data handling SIM for use with the MMS is economically desirable and should be undertaken. However, the study further indicates that significantly greater cost savings can be realized by the extension of the SIM concept to the Spacelab and other unmanned spacecraft missions. For this reason, NASA is urged to consider the requirements for the use of SIM, as well as the economic effects of standardization across the entire mission model, as opposed to considering the use of SIM with only the MMS. This further study could possibly lead to the identification of additional types of SIM, as well as to an improved basis for economic justification of the SIM concept. When further study of this concept is undertaken, specific areas of uncertainty, such as the optimum level of modularity, spares philosophy and costs of ground support SIM, should be resolved.